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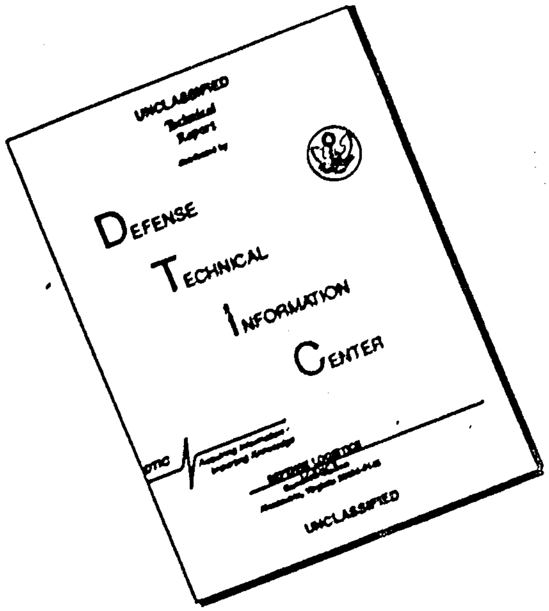

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Pratt & Whitney Aircraft DIVISION OF UNITED AIRCRAFT CORPORATION

December 27, 1967

In reply please refer to:
MFS:RPSg:Cont. Adm.

Captain Ernie D. Braunschweig, RPRZ
Air Force Rocket Propulsion Laboratory
Edwards, California 93523

Dear Captain Braunschweig:

Per P&WA letter, MFS:RPSg:Cont. Adm., dated November 22, 1967, we transmitted AFRPL-TR-67-270, Applications Study for a High Performance Cryogenic Staged Combustion Rocket Engine - Final Report, classified "Confidential."

Subsequent to the transmittal of this report, certain information contained therein and originally designated as unclassified has been classified "Confidential." We therefore request that you take the necessary action to mark your copy of this document as indicated below so that viewers will be properly advised.

The following pages of AFRPL-TR-270 are classified "Confidential."

(a) Page 73 - Figure 80
(b) Page 148 - Paragraph 1 and Figure 155
(c) Page 149 - Paragraphs 2, 3, 5 and 6
(d) Page 181 - Section 4.2.1 (A & B)
(e) Page 273 - Ground Rule No. 10

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Very truly yours,

UNITED AIRCRAFT CORPORATION
Pratt & Whitney Aircraft Division

M. F. Samples

M. F. Samples
Contract Administrator
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(UNCLASSIFIED TITLE)

APPLICATIONS STUDY FOR A HIGH PERFORMANCE CRYOGENIC STAGED COMBUSTION ROCKET ENGINE

FINAL REPORT

R. R. ATHERTON

FOREWORD

This final report describes the Applications Study for A High Performance Cryogenic Staged Combustion Rocket Engine conducted during the period 1 March 1966 to 31 July 1967, and is submitted in accordance with the requirements of Contract AF 04(611)-11401.

This publication was prepared by the Pratt & Whitney Aircraft Florida Research and Development Center as report PWA FR-2471.

Classified information has been extracted from (asterisked) documents listed under References.

This Technical Report has been reviewed and is approved.

Robert E. Probst
Captain, USAF
Project Engineer
Air Force Rocket Propulsion Laboratory

Ernie D. Braunschweig
Captain, USAF
Program Manager
Air Force Rocket Propulsion Laboratory

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UNCLASSIFIED ABSTRACT

The Applications Study was conducted as a part of the Advanced Development Program for a High Performance Staged Combustion Oxygen/Hydrogen Rocket Engine. The overall objective of the Applications Study was to investigate the application of this engine in six representative advanced rocket stages and to determine the resulting performance. A stage figure of merit (referred to as the Performance Index, W_x) was specified and used for performance evaluation. Performance Index included considerations of engine specific impulse, weight, size, and installation features as these parameters affect vehicle performance. Engine associated weight includes items such as thrust structure, feed lines, thrust vector control, failure detection equipment, and propellant tank pressurization. The analysis was conducted at 250K and 350K vacuum thrust levels in each of six vehicle applications.

Based upon the results of the Applications Study, the lightweight two-position bell nozzle was selected as the basic engine nozzle configuration. Engine nozzle expansion ratio varied considerably in each of the six vehicle cases. Lower stages tended to optimize with lower expansion ratio nozzles and upper stages with higher expansion ratios. A common engine module size for both the 250K and 350K thrust levels was determined by collectively considering the requirements for all six vehicles. Final Performance Index values were determined using this common engine module and exhaust nozzles individually selected (expansion ratio, contour, fixed or two-position) to provide best vehicle performance. The final summed performance of the six vehicle cases (250K engine) was 98.2% of the performance that could be realized for cases where the engine module was individually sized in each vehicle case.

A complete engine description including parametric data and operating parameters is included in the report.

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TABLES

TABLES (Continued)

LIST OF ABBREVIATIONS AND SYMBOLS

ITEM	DEFINITION
A_f	Fairing sufrace, area, ft^2
A_i	Interstage surface area, ft^2
Base	Base nozzle truncation
F_{vac}	Vacuum thrust, lb
h	Thermal conductivity, Btu/ft^2hr°R
I	Moment of inertia, slug/ft^2
I_s	Specific impulse, sec
I_s	Sea level specific impulse, sec
$I_{s_{alt}}$	Specific impulse at the altitude of interest, sec
I_{vac}	Vacuum specific impulse, sec
K	Stage weight-growth constant
L	Trajectory loss parameter
$\mathcal{M}$	Molecular weight
MC_S	Maximum performance nozzle truncation
MH	Engine mount height, in.
ML	Minimum length nozzle truncation
MR	Engine mount ring radius, in.
MSA	Minimum surface area nozzle truncation
NPSH	Net positive suction heat, ft
P_a	Ambient pressure, psia
P_c	Chamber pressure, psia
r	Mixture ratio
SITVC	Secondary injection thrust vector control
T_T	Total temperature, °R
TBO	Time between overhaul
TVC	Thrust vector control
$\dot{w}$	Total propellant flow rate per engine, lb/sec/engine
Wbc	Corrected stage burnout weight, lb
We	Installation weight, lb
Wg	Stage gross weight, lb
W_x	Performance index, lb

LIST OF ABBREVIATIONS AND SYMBOLS (Continued)

ITEM	DEFINITION
ϵ	Expansion ratio
P	Primary or retracted two position nozzle
γ	Ratio of specific heat
V	Ideal velocity increment, ft/sec

SECTION 1
INTRODUCTION

(U) This final report summarizes the results of the Applications Study portion of the Advanced Development Program for a High Performance Cryogenic Rocket Engine. The overall purpose of the analysis was to evaluate performance of the high-pressure, bell-nozzle engine concept in six different propulsion applications. A secondary objective was to evaluate vehicle performance improvements possible with improvements in engine performance or different operating modes. A specified general approach and a figure of merit (Performance Index) were used throughout the study of the six specified vehicle configurations.

(C) Six vehicle cases were studied for 250K and 350K vacuum thrust engine sizes. The six cases specified for this study were:

Case 1 - expendable lower stage
Case 2 - expendable upper stage
Case 3 - expendable single-stage to orbit
Case 4 - recoverable lower stage
Case 5 - recoverable upper stage, pick-a-back
Case 6 - recoverable upper stage, tandem.

These application cases are shown schematically for both engine sizes in figures 1 and 2.

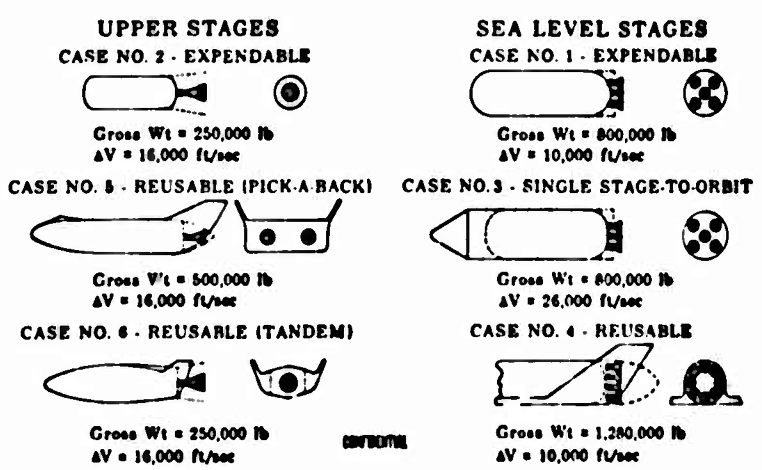

Figure 1. Application Study Vehicles, 250K Engines FD 21995

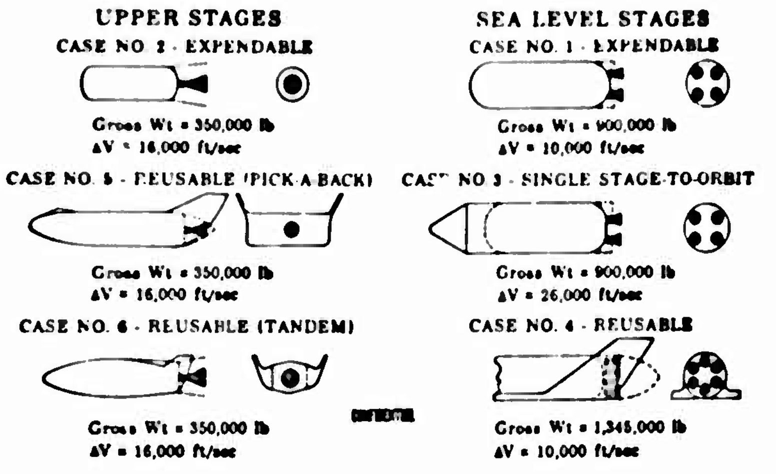

Figure 2. Application Study Vehicles, 350K Engines

FD 21996

(U) The figure of merit used in the Application Study is the Performance Index (W_x) and is defined in equation form in the Air Force supplied application package (Appendix I). The Performance Index calculation includes parameters directly associated with the vehicle application along with engine installation size and weight parameters. This equation parametrically considers the interaction of engine performance, size, and location with overall vehicle performance. Vehicle requirements and trajectory parameters were specified, however, all equipment associated with the engine installation were analyzed. This included trade-off studies of individual sub-component systems along with evaluation within the overall system and application.

(U) A single common module was defined by detailed optimization procedures for all of the application cases. For the purpose of this study a common module was defined as a basic engine power package in which the flow rate and chamber pressure remained constant for all six vehicle applications. The high pressure engine concept permits the same engine power package to be used with nozzle skirts of different area ratio; each skirt is therefore tailored to provide optimum performance for a specific application case.

(U) Engine configurations evaluated included the two-position nozzle concept. With this concept, the nozzle skirt is retracted to reduce engine length and increase sea level thrust and specific impulse.

SECTION II
SUMMARY

(U) The applications study was conducted as a continuing effort throughout the Phase I Advanced Development Program. Performance Index (W_x) values have been updated as new or improved engine and engine-vehicle interface data became available. Initial work was done with high pressure engines having fixed regeneratively cooled exhaust nozzles, however, the final W_x values summarized in this report (with the exception of Case 4) are for engines using lightweight two-position nozzles. Case 4, because of severe diameter constraints and resulting low area ratio requirements, shows slightly better performance with a fixed nozzle. All other cases show a W_x improvement for the lightweight two-position nozzle concept.

(C) Tables I and II provide the final W_x values obtained by using the common expansion ratio selection technique as discussed later and shown on bar No. 2 of figure 3. These performance data are based on lightweight two-position nozzle engines, except for Case 4, which used a fixed nozzle. The data for the 250K application cases are summarized in table I and table II is the summary for the 350K cases. An expansion ratio of 250 was selected for the basic 250K module and an expansion ratio of 300 was selected for the 350K module. Other pertinent engine and engine-vehicle interface data are also shown. The application cases require a wide range of engine mixture ratio as shown by the optimum values for each case in tables I and II.

(U) A requirement of this study was to establish a common engine module (or power package) for use in the six vehicle cases. The exhaust nozzle used with this common module can vary in area ratio (or configuration) to produce optimum stage performance with the limitation that no engine configuration exceed 250K (350K) vacuum thrust. With the high-pressure bell-nozzle engine concept, different nozzles can be used with the same basic engine module. For this study a common module was defined as a basic engine power package in which propellant flow rate and chamber pressure characteristics as a function of mixture ratio remains constant for all six vehicle cases. Engine thrust will vary with the different nozzles, however, as noted above vacuum thrust does not exceed 250K (or 350K) for any engine configuration.

(C) It was found that Performance Index (W_x) values were sensitive to stage thrust since a constant gross weight was imposed by the vehicle modules. This sensitivity occurs because of vehicle gravity and drag losses and was especially strong in the lower stages. Performance Index values were thus dependent upon the technique used to establish the basic module size. Engine thrust (with a constant nozzle expansion ratio) is a function of propellant flow rate, i.e., thrust increases with increasing propellant flow rate. This effect of propellant flow rate on W_x is shown in figure 4. For any given engine expansion ratio there is a specific module flow rate that will produce 250K vacuum thrust. At flow rates higher than this level, higher vacuum thrust are produced. In figure 4 the vehicle application case lines are dotted for propellant flow rates that would produce more than 250K thrust (with the nozzle expansion ratio found optimum for that stage). The vertical dotted line at a propellant flow rate of 536.4 lb/sec represents the module flow rate selected in the final optimization.

(C) Table I. Performance Index Summary (250K Study)

	Case 1	Case 2	Case 3	Case 4	Case 5	Case 6
Performance Index, W_x (lb)	235,600	66,350	45,180	306,100	80,850	39,350
Engine Description						
Vacuum Thrust, F_{vac} (lb/engine)	243,400	250,000	245,700	237,000	248,500	250,000
Number of Engines	5	1	5	8	2	1
Chamber Pressure, P_c (psi)	3,000	3,000	3,000	3,000	3,000	3,000
Engine Module Size, ϵ_M*	250	250	250	250	250	250
Nozzle Contour	MC_8	MSA	MC_s	MC_8	MSA	MSA
Expansion Ratio, ϵ (Retracted)	35	111	35		89	111
(Translated)	75	250	110	35	200	250
Mixture Ratio, r	5.15	5.9	5.76	5.52	6.4	6.7
Vacuum Specific Impulse, I_{vac} (sec)	455.2	466.3	458.7	443.0	461.3	462.0
Sea Level Specific Impulse, I_{sl} (sec)	396.8	—	389.0	403.2	—	—
Exit Diameter, OD (in.)	64.5	116.7	78.0	44.3	104.5	116.7
Engine Length (in.) (Retracted)	91.3	104.0	122.5		86.5	104.0
(Extended)	143.2	194.0	170.0	104.0	173.0	194.0
Engine Mount Height, MH (in.)	88	73	80	47	52	45
Engine Mount Ring Radius, MR (in.)	101	—	118	98	—	—
Installation Weight, W_e (lb)	18,361	4474	19,258	26,141	8857	4189
Oxidizer Feed Lines (lb)	383	55	385	417	143	55
Fuel Feed Lines (lb)	239	55	300	551	136	55
Thrust Structure (lb)	2190	1285	2610	2335	2500	1000
Heat Shield (lb)	762	—	993	725	—	—
TVC System (lb)	1450	290	1450	2320	580	290
Pressurization System (lb)	152	24	145	233	48	24
Failure Detection System (lb)	275	55	275	440	110	55
Engine(s) (lb)	12,850	2710	13,100	19,120	5340	2710
Fairing Area, AF (ft^2)	711	—	776	1437	2160	412
Interstage Area, A_I (ft^2)	—	977	—	—	—	356

*The area ratio required for 250K vacuum thrust using one common set of p[illegible]burner, turbopumps, and main chamber hardware.

(C) Table II. Performance Index Summary (350K Study)

	Case 1	Case 2	Case 3	Case 4	Case 5	Case 6
Performance Index, W_x (lb)	265,900	95,230	53,950	325,000	51,040	60,800
Engine Description						
Vacuum Thrust, F_{vac} (lb/engine)	340,700	350,000	344,400	330,500	343,800	350,000
Number of Engines	4	1	4	6	1	1
Chamber Pressure, P_c (psi)	3,000	3,000	3,000	3,000	3,000	3,000
Engine Module Size, ϵ_M*	300	300	300	300	300	300
Nozzle Contour	MC_s	MSA	MC_s	MC_s	MSA	MSA
Expansion Ratio, (Retracted)	35	140	35		70	140
(Translated)	80	300	135	35	150	300
Mixture Ratio, r	5.00	5.85	5.82	5.48	6.5	6.8
Vacuum Specific Impulse, I_{vac} (sec)	456.4	468.5	461.0	443.5	457.5	463.8
Sea Level Specific Impulse, I_{sl} (sec)	395.2	—	387.5	403	—	—
Exit Diameter (in.)	79.2	151.0	101.5	52.1	107	151.0
Engine Length (in.) (Retracted)	114.1	133.5	169.5		98.6	133.5
(Translated)	170.0	250.0	219.0	116.5	173.5	250
Engine Mount Height, MH (in.)	78	78	60	60	102.5	60
Engine Mount Ring Radius, MR (in.)	100	—	120	88	—	—
Installation Weight, W_e (lb)	18,767	5903	19,684	26,231	4998	5515
Oxidizer Feed Lines (lb)	345	66	336	550	61	71
Fuel Feed Lines (lb)	360	64	398	530	194	71
Thrust Structure (lb)	1930	1730	2210	2720	860	1330
Heat Shield (lb)	780	—	1070	655	—	—
TVC System (lb)	1360	340	1360	2040	340	340
Pressurization System (lb)	152	28	150	236	28	28
Failure Detection System (lb)	220	55	220	330	55	55
Engine(s) (lb)	13,620	3620	13,940	19,170	3460	3620
Fairing Area, AF (ft^2)	696	—	698	1515	2506	539
Interstage Area, A_i (ft^2)	—	1219	—	—	—	536

*The area ratio required for 350K vacuum thrust using one common set of preburner, turbopumps, and main chamber hardware.

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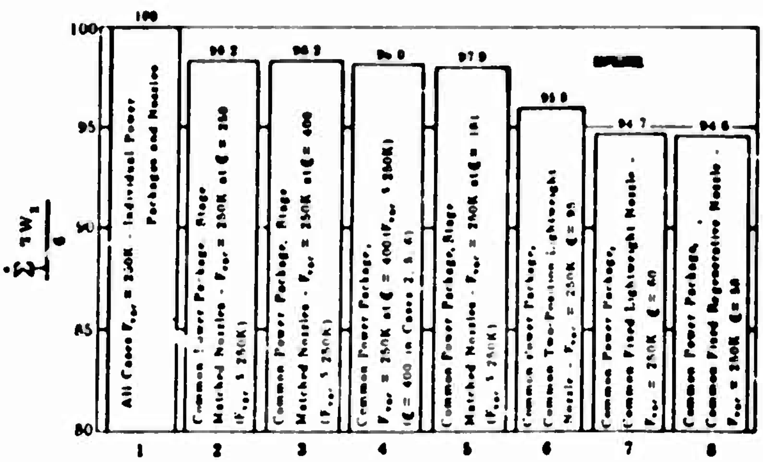

Figure 3. Summary of Alternative Module Sizing (250K Module) FD 23110

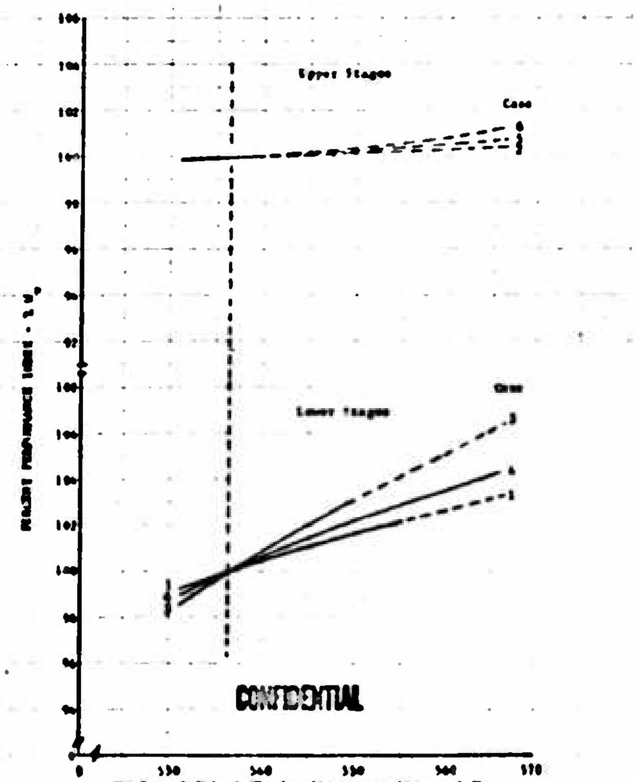

Figure 4. Percent Performance Index (250K Module) DF 57576

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(C) Because of the effect of propellant flow rate on W_x, it was decided to investigate a number of optimization techniques in addition to the one suggested in the Air Force supplied Applications Package. The results of this study in terms of a normalized summation of W_x values are shown in figure 3. The first bar (on the left of the figure) is used as a standard (100%) and is obtained by optimizing each of the six cases parametrically with engines having 250K vacuum thrust at all points. The recommended technique (and used for the final W_x values in this report) is shown on the second bar. This bar represents a composite summation of all six vehicle cases to establish the expansion ratio for 250K vacuum thrust and, thus, common module size. No engine was allowed to exceed an area ratio of 250, thus no engine exceeds 250K vacuum thrust; however, higher performance could be obtained with this technique if Cases 2 and 6 were allowed to go to the stage matched optimum area ratio of 400. The data used to generate the performance shown in figure 3 are discussed in detail in Section V.

(C) A supplementary study was conducted to determine if vehicle performance improvements (W_x) could be obtained by selecting other than a constant engine mixture ratio during vehicle flight. It was postulated that operation of the engine at high mixture ratios (and lower specific impulse) during the initial portion of the flight trajectory would lighten the vehicle more rapidly and reduce gravity and drag losses. Lower than normal mixture ratio (higher specific impulse) would be reserved for the later portion of the flight. It was determined that programming the mixture ratio did not improve vehicle performance. A similar investigation was made utilizing the capability of the engine to operate at slightly higher thrust levels at mixture ratios other than 5.0 or 7.0. This study showed that vehicle performance could be improved by operating the engine at the slightly higher thrust and higher mixture ratios in the early thrust sensitive part of the flight and then shifting to a lower mixture ratio later in the flight. If this mode of operation were used, a new engine control schedule would be required because the present schedule holds thrust constant at mixture ratios from 5 to 7.

(U) A study was conducted to determine if thrust vector control by addition of a secondary fluid in the exhaust nozzle to create a differential pressure field was competitive with mechanical gimbaling. It was determined that mechanical gimbaling was clearly superior in all six vehicle cases.

SECTION III
MODULE ENGINE DESCRIPTION AND PERFORMANCE

(U) The engine module used in the application study is a high performance oxygen/hydrogen rocket engine. This section provides a functional description of the engine concept and provides performance for the specific modules selected from the applications study optimization analysis.

A. GENERAL DESCRIPTION

(C) The staged-combustion, high-pressure engine with a two-position nozzle is a versatile, compact, high-performance propulsion system for use in both upper and lower stages of the advanced vehicles considered in the applications study. This reusable engine is capable of maintaining constant thrust over a mixture ratio range of 5 to 7. Nozzle interchangeability and the two-position nozzle concept permit operation of the same engine system with optimum nozzle area ratios for improving the performance of the lower stages within the atmosphere; and providing the high performance attainable with very high area ratio nozzles in the upper stages. This interchangeability is achieved by having a constant turbomachinery power package and attaching the desired nozzle skirt for the various application requirements. The area ratio, at which the fixed (primary) nozzle ends and the translating (secondary) skirt starts, is varied to optimize the performance for the specific application study case. An isometric cut-away of the basic engine with a lower stage two-position nozzle is shown in figure 5.

(C) A propellant flow schematic illustrating the principal flow paths and the functional component arrangement of this engine is shown in figure 6. Hydrogen and oxygen enter at the engine-driven low-speed inducers. The low-speed inducers are used to minimize vehicle tank pressure requirements while maintaining high-speed main propellant pumps for high turbopump efficiencies. The low-speed fuel inducer is a single shaft unit with a high specific speed, axial-flow inducer driven by a partial-admission, single-stage, hydrogen turbine. The low-speed oxygen inducer is also a single shaft unit with a high specific speed, axial-flow inducer driven by a partial admission, single-stage liquid oxygen turbine.

(C) The main fuel turbopump is a single shaft unit with two, back-to-back, centrifugal pump stages driven by a two-stage, pressure-compounded turbine. A double acting thrust balance piston is provided between the pump and turbine.

(C) The oxidizer turbopump is a single shaft unit with a single, shrouded centrifugal pump stage driven by a two-stage, pressure-compounded turbine. A single-acting thrust balance piston is provided between the pump and turbine.

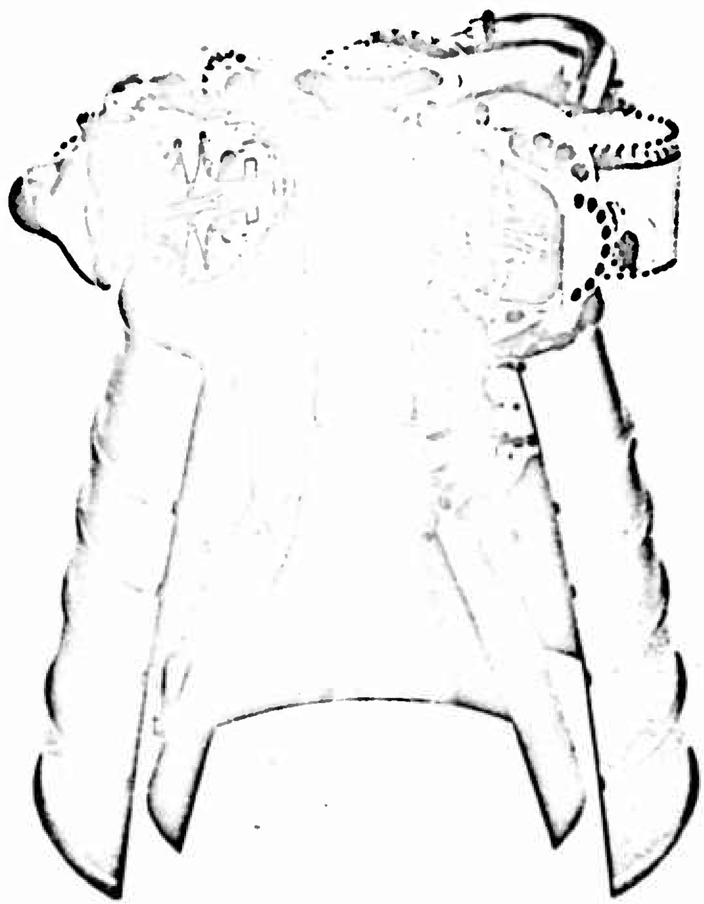


Figure 5. 250,000-lb Thrust Engine FD 21027B

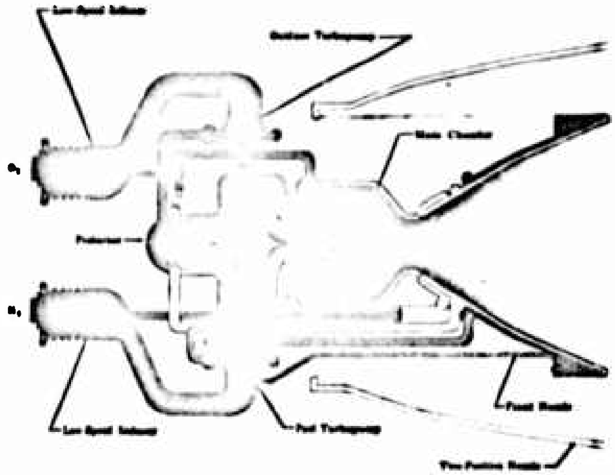


Figure 6. Propellant Flow Schematic FD 210028

(C) The preburner injector consists of dual-orifice tangential-swirler oxidizer injector elements and variable-area fuel injector elements. A combination liquid oxygen throttle valve and flow divider valve is incorporated at the rear of the injector assembly to vary the total oxidizer flow rate for turbine inlet temperature control, to adjust the relative flow of the primary and secondary elements, and adjust engine power level. The preburner combustion chamber is an integral part of the transition case, which contains the turbine drive gas ducts and a cooled outer shell. The main turbopumps are mounted to the transition case with a plug-in arrangement of the turbines for maintainability.

(C) The main chamber injector consists of fixed-area, tangential-swirler oxygen injection elements arranged in radial spraybars. The fuel side (preburner combustion products after expansion through the turbine) is a fixed-area design that directs fuel-rich gas flows through a porous faceplate to provide cooling. The combustion chamber wall is composed of a hydrogen cooled liner extending from the injector face through the throat region to a point immediately downstream of the throat. The liner is composed of porous plates providing transpiration cooling.

(C) The nozzle, which attaches immediately downstream of the throat, is composed of a regeneratively cooled section and a low-pressure, dump-cooled section. The forward portion, which is in the higher heat flux region, is regeneratively cooled with conventional tubular heat exchangers.

(C) The main hydrogen flow is pumped to system operating pressure levels by the main fuel pump and the hydrogen is then ducted to cool the regenerative section of the nozzle. A portion of the fixed nozzle at the rear of the regeneratively cooled section is cooled with the total hydrogen flow from the pump in a single pass, counter-flow heat exchanger. Most of this flow exits the nozzle at a downstream point and is ducted to

the preburner. The remainder, a small portion of the hydrogen, is used to regeneratively cool the smaller forward section of the regeneratively cooled nozzle and is subsequently used as a working fluid to power the fuel low-speed inducer drive turbine. This flow is then used in the transpiration cooled main chamber walls. Because this flow rate is greater than required for transpiration cooling, a portion is returned to the system at a point downstream of the turbines.

(C) A small amount of hydrogen is used at low supply pressure to cool the nozzle skirt downstream of an expansion ratio of 35. This hydrogen is heated to high temperature in the skirt and expelled overboard through small nozzles at the end of the skirt. (The low pressure nozzle skirt cooling flow is ducted from the discharge of the fuel low-speed inducer.) A quick-disconnect fitting is provided to stop the flow when the secondary nozzle is retracted.

(C) After being pumped to system operating pressure, the oxygen is divided between the preburner and the main chamber. The smaller portion of the flow is supplied to the preburner and is burned with the hydrogen. The resulting combustion products provide the working fluid for the main turbines, which are arranged in parallel. The turbine exhaust gases are rejoined and directed to the main injector.

(C) The main chamber oxygen flow provides the oxygen low-speed inducer turbine working fluid and uses the available pressure drop between the main oxidizer pump discharge pressure and the main chamber for turbine power. The oxidizer flow is then injected into the main combustion chamber and is mixed and burned with the fuel-rich turbine exhaust gases. The resulting combustion gas is then expanded through the bell nozzle.

(C) The primary engine thrust and mixture ratio controls are located in the liquid oxygen supply lines to the preburner and the main chamber and in the variable-area fuel side of the preburner injector.

B. PERFORMANCE

(U) The complete set of engine parametric data used as the basis for the applications study is given in Appendix II.

1. 250K Module

(C) The complete engine installation optimization analysis of the six 250K module application study cases resulted in the selection of a common engine module developing 250,000 pounds of vacuum thrust at an overall nozzle expansion ratio of 250. The bell nozzle is a minimum surface area contour and translates from an area ratio of 111. A layout of the common engine is shown in figure 7.

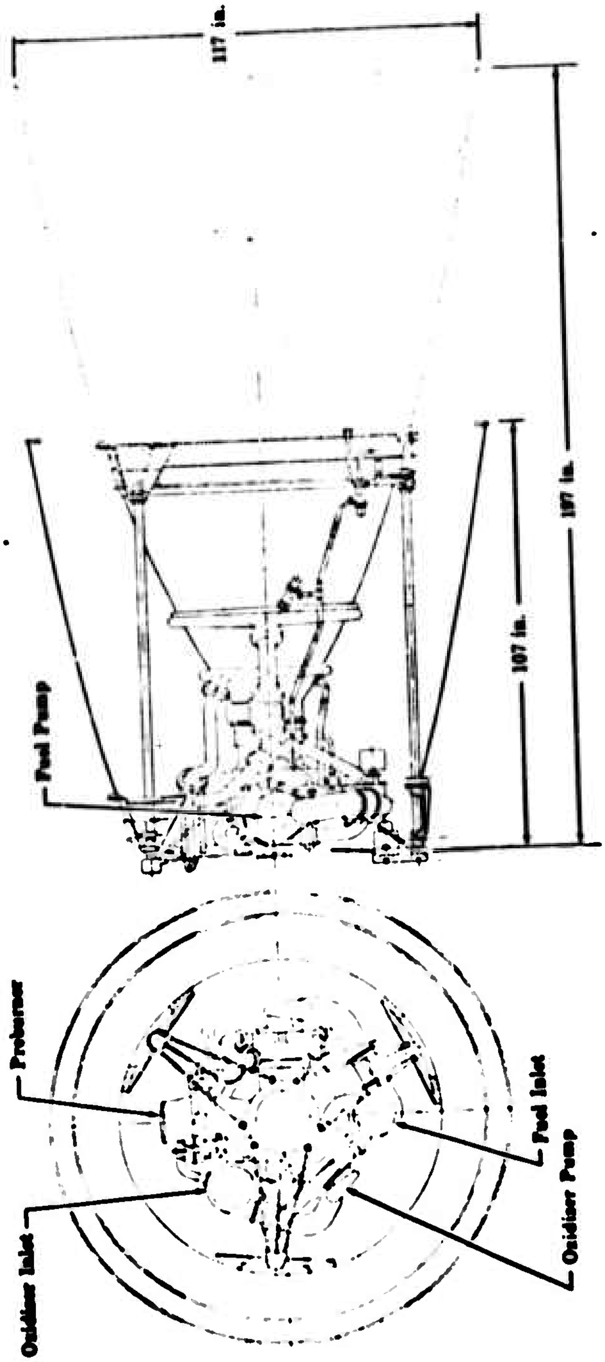


Figure 7. High Pressure Engine Installation Drawing (250K Module)

(C) The engine characteristics are based on the following:

1. High pressure, staged-combustion, two-position bell nozzle engine
2. Performance available at an engine preflight rating test (PFRT) (Flight Engine Performance Level.)
3. Engine inlet propellant conditions of
    a. Minimum hydrogen net positive suction head (NPSH) = 60 ft
    b. Minimum oxygen net positive suction head (NPSH) =.16 ft
4. Continuous throttling capability between 100% and 20% of rated thrust
5. Mixture ratio range of 5 to 7 at all thrust levels
6. Thrust vector control provided by mechanical gimbaling
7. Durability of 10 hours time between overhaul (TBO), 100 reuses, 300 starts, 300 thermal cycles, and 10,000 valve cycles
8. Lightweight two-position nozzle that translates to provide high sea level and altitude performance, in addition to compact packaging
9. Performance is based on the use of dump cooling downstream of an expansion ratio of 35
10. For high expansion ratio nozzles, radiation cooling is used aft of the lowest expansion ratio permitted by heat flux levels. (This expansion ratio varies over the parametric range, but is approximately 200.) If radiation cooled nozzle skirts were not used, an insignificant increase in engine weight would result.

(U) A table of engine operating parameterr for the 250K common module at the extremes of the operating range of thrust and mixture ratio is shown in table III.

(U) Using the basic propellant flow rate characteristics, nozzle expansion ratios were selected for the requirements of each application case. Table IV presents relevant engine data. Nozzle contour is also indicated. A complete weight breakdown of the components in the power package is provided in table V.

(C) Table III. Operating Characteristics of Common Module Engine

	100% Thrust r = 5	100% Thrust r = 6	100% Thrust r = 7	20% Thrust r = 5	20% Thrust r = 6	20% Thrust r = 7
Vacuum Thrust, lb	250,000	250,000	250,000	50,000	50,000	50,000
Mixture Ratio	5.0	6.0	7.0	5.0	6.0	7.0
Vacuum Specific Impulse, sec	467	466	460	466	463	456
Overall Specific Impulse Efficiency	0.974	0.969	0.961	0.974	0.968	0.960
Nozzle Area Ratio						
Extended	250					
Retracted	111					
Overall Diameter, in.	116.7					
Length, in.						
Extended	194					
Retracted	104					
Weight, lb.	2710					
Main Combustion Chamber and Nozzle						
Chamber Pressure, psia	3090	3000	2925	615	595	575
Injector Mixture Ratio	5.3	6.4	7.5	5.3	6.5	7.6
Transpiration Coolant Flow, lb/sec	2.3	[illegible].5	2.6	0.6	0.6	0.7
Nozzle Skirt Coolant Flow, lb/sec	2.5	2.4	2.2	0.5	0.5	0.4

(C) Table III. Operating Characteristics of Common Module Engine (Continued)

	100% Thrust r = 5.	100% Thrust r = 6.	100% Thrust r = 7.	20% Thrust r = 5	20% Thrust r = 6	20% Thrust r = 7
Preburner						
Total Propellant Flow, lb/sec	156	140	127	24	22	20
Mixture Ratio	1.0	1.2	1.3	0.7	0.9	1.0
Combustion Temperature, °R	1844	2103	2324	1270	1590	1840
Combustion Pressure, psia	5036	4748	4500	820	795	765
Combustion Efficiency	0.983	0.983	0.983	0.983	0.983	0.983
Fuel Turbopump						
Pump						
Flow Rate, lb/sec	8[illegible].5	74.1	65.8	17.4	14.9	13.3
Speed, rpm	48,000	47,700	46,700	22,600	23,600	23,700
Pressure Rise, psi	5520	5720	5540	1225	1300	1280
Efficiency	0.718	0.710	0.700	0.562	0.511	0.480
Turbine						
Flow Rate, lb/sec	102	92	83	16	14	14
Inlet Temperature	1844	2103	2324	1269	1589	1841
Pressure Ratio	1.5	1.5	1.4	1.3	1.3	1.2
Efficiency	0.742	0.742	0.741	0.682	0.676	0.672

(C) Table III. Operating Characteristics of Common Module Engine (Continued)

	100% Thrust r = 5	100% Thrust r = 6	100% Thrust r = 7	20% Thrust r = 5	20% Thrust r = 6	20% Thrust r = 7
Oxidizer Turbopump						
Pump						
Flow Rate, lb/sec	446.0	459.5	475.0	89.1	92.6	95.8
Speed, rpm	27,100	25,600	23,900	11,770	11,750	11,430
Pressure Rise, psi	6750	5950	5050	1380	1380	1305
Efficiency	0.722	0.728	0.730	0.496	0.507	0.527
Turbine						
Flow Rate, lb/sec	54	49	45	8	8	7
Inlet Temperature, °R	1844	2103	2324	1269	1589	1841
Pressure Ratio	1.4	1.4	1.3	1.2	1.2	1.2
Efficiency	0.720	0.715	0.707	0.622	0.601	0.587
Fuel Low-Speed Inducer						
Inducer						
Flow Rate, lb/sec	89.0	76.5	68.0	17.9	15.4	13.7
Speed, rpm	20,500	19,300	18,000	6560	6700	6290
NPSH, ft	60	60	60	60	60	60
Pressure Rise, psi	51	68	68	16	19	18
Efficiency	0.792	0.778	0.764	0.473	0.407	0.378
Turbine						
Flow Rate, lb/sec	4.3	5.0	4.9	1.1	1.3	1.3
Pressure Ratio	1.3	1.3	1.3	1.3	1.4	1.4
Efficiency	0.511	0.486	0.477	0.193	0.194	1.191

(C) Table III. Operating Characteristics of Common Module Engine (Continued)

	100% Thrust r = 5	100% Thrust r = 6	100% Thrust r = 7	20% Thrust r = 5	20% Thrust r = 6	20% Thrust r = 7
Oxidizer Low-Speed Inducer						
Inducer						
Flow Rate, lb/sec	446.0	459.5	475.0	89.1	92.6	95.8
Speed, rpm	5480	5400	5420	1975	1965	1890
NPSH, ft	16	16	16	16	16	16
Pressure Rise, psi	175	160	150	35	35	30
Efficiency	0.791	0.805	0.806	0.386	0.400	0.425
Turbine						
Flow Rate, lb/sec	369	387	406	80	83	86
Pressure Drop, psi	578	533	514	245	225	175
Efficiency	0.457	0.443	0.433	0.404	0.416	0.440

(C) Table IV. Engine Performance Summary (250K Module)

	Basic Module	Case 1	Case 2	Case 3	Case 4	Case 5	Case 6
Vacuum Thrust, lb	250,000	243,400	250,000	245,700	237,000	248,500	250,000
Chamber Pressure, psia (Nominal)	3,000	3,000	3,000	3,000	3,000	3,000	3,000
Nozzle Expansion Ratio	250	75	250	110	35	200	250
Primary Nozzle Expansion Ratio	111	35	111	35		89	111
Nozzle Contour Type	MSA	MC_s	MSA	MC_s	MC_s	MSA	MSA
Case Nominal Mixture Ratio, r	—	5.15	5.90	5.76	5.52	6.4	6.7
Vacuum Specific Impulse, sec	—	455.2	466.3	458.7	443	461.3	462.0
Sea Level Specific Impulse, sec	—	396.8	—	389	401.2	—	—
Specific Impulse at Transition Altitude, sec	—	424	—	416	—	—	—
Transition Altitude, ft	—	25,000	—	28,200	—	—	—
Exit Diameter, in.	116.7	64.5	116.7	78.0	44.3	104.5	116.7
Extended Length, Gimbal Axis to Exit Plane, in.	194.0	143.2	194.0	170.0	104.0	173.0	194.0
Stowed Length, in.	104.0	91.3	104.0	122.5	—	93.7	104.0
Power Package Weight, lb	2230	2230	2230	2230	2230	2230	2230
Nozzle and Two-Position Actuator Weight, lb	480	340	480	390	160*	440	480
Engine Total Weight, lb	2710	2570	2710	2620	2390	2670	2710

* Nozzle Only

(C) Table V. Power Package Component Weights

	250K Module	350K Module
Fuel Turbopump	270	370
Fuel Low-Speed Inducer Turbopump	100	110
Oxidizer Turbopump	250	340
Oxidizer Low-Speed Inducer Turbopump	115	130
Plumbing	110	145
Controls and Valves	320	405
Transition Case	280	385
Preburner Injector and End Cap	275	275
Combustion Chamber	270	340
Main Injector	125	170
Gimbal	40	55
Miscellaneous Hardware	75	100
TOTAL	2230	2825

(U) Vacuum specific impulse is shown in figure 8 as a function of mixture ratio for the selected nozzle expansion ratios. Altitude performance for the lower stage cases is shown in figure 9. The optimum altitude for secondary nozzle translation is indicated by the change in slope. A complete range of performance, weight, and dimensional characteristics for the selected common 250K module is given in Appendix III as a function of nozzle configuration and mixture ratio.

2. 350K Module

(C) The selected compromise engine is a module developing 350,000 pounds of vacuum thrust at an overall nozzle expansion ratio of 300. Table VI is a summary of the engine data for the various cases. The power package component breakdown is presented in table V. Figure 10 shows vacuum performance and the altitude performance is shown in figure 11. A complete range of performance, weight, and dimensional data for the selected 350K common module are presented in Appendix III.

(U) The engine characteristics for the 350K module are similar to those provided for the 250K module.

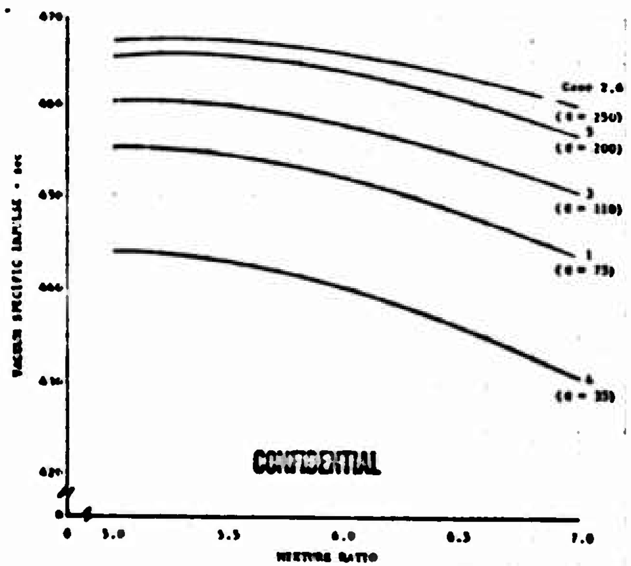


Figure 8. Vacuum Specific Impulse vs Mixture Ratio (250K Module) DF 57589

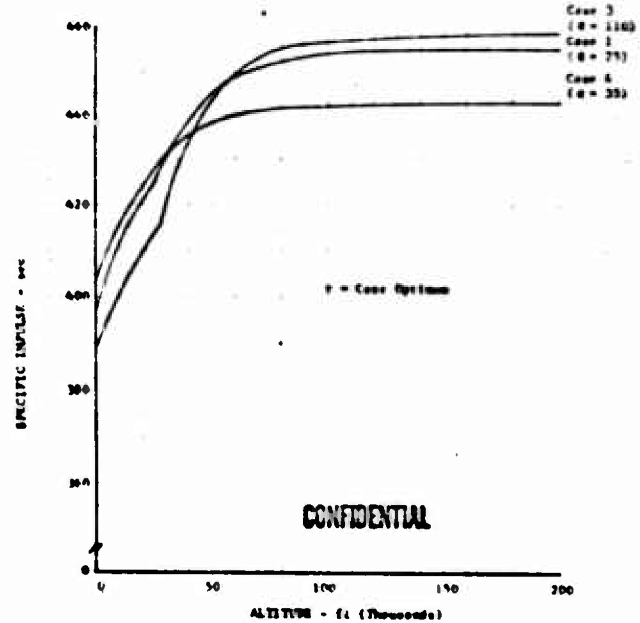


Figure 9. Altitude Performance for 250K Module DF 57570

(C) Table VI. Engine Performance Summary (350K Study)

	Basic Module	Case 1	Case 2	Case 3	Case 4	Case 5	Case 6
Vacuum Thrust, lb	350,000	340,700	350,000	344,400	330,500	343,800	350,000
Chamber Pressure, psia (Nominal)	3,000	3,000	3,000	3,000	3,000	3,000	3,000
Nozzle Expansion Ratio	300	80	300	135	35	150	300
Primary Nozzle Expansion Ratio	140	35	140	35	—	70	140
Nozzle Contour Type	MSA	MC_s	MSA	MC_s	MC_s	MSA	MSA
Engine Mixture Ratio, r	5-7	5.0	5.85	5.82	5.48	6.5	6.8
Vacuum Specific Impulse, sec	—	456.4	468.5	461.0	443.5	457.5	463.8
Sea Level Specific Impulse, sec	—	395.2	—	387.5	403.0	—	—
Specific Impulse at Transition Altitude, sec	—	421.7	—	415.5	—	—	—
Transition Altitude, ft	—	26,200	—	30,500	—	—	—
Outside Exit Diameter, in.	151.0	79.2	151.0	101.5	52.1	107	151.0
Extended Length, Gimbal Axis to Exit Plane, in.	250.0	170.0	250.0	219.0	116.5	173.5	250.0
Stowed Length, in.	133.5	114.1	133.5	169.5	—	98.6	133.5
Power Package Weight, lb	2825	2825	2825	2825	2825	2825	2825
Nozzle and Two-Position Actuator Weight, lb	795	580	795	660	370*	635	795
Engine Total Weight, lb	3620	3405	3620	3485	3195	3460	3620

* Nozzle only

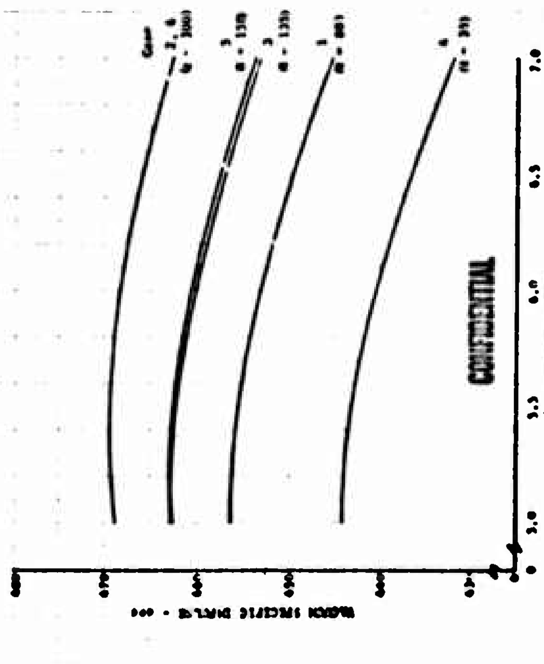


Figure 10. Vacuum Specific Impulse vs Mixture Ratio (350K Module) DF 57655

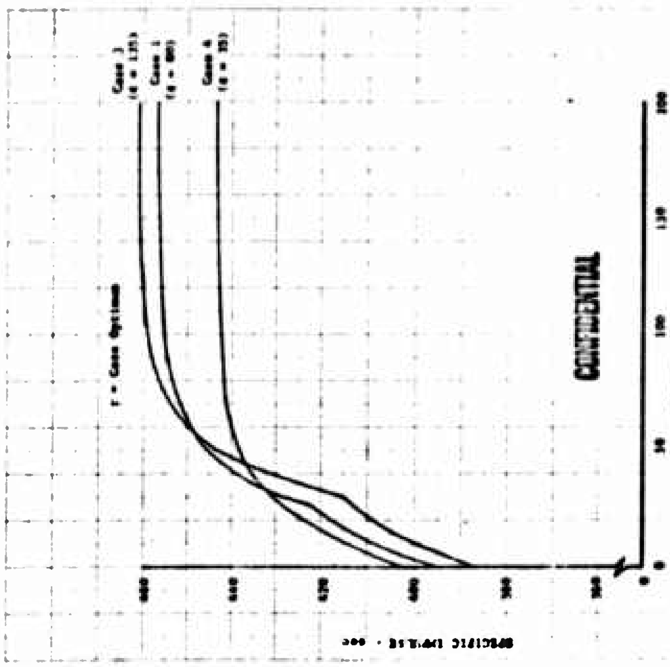


Figure 11. Altitude Performance for 350K Module DF 57654

SECTION IV
OPTIMIZATION STUDIES

A. PERFORMANCE INDEX

(U) The Performance Index equation and procedures outlined in the vehicle applications package were used to evaluate engine performance with realistic vehicle, mission, and installation constraints. The vehicle application package procedure was also used to determine the best common module engine and to optimize stage (six applications) performance with this module. This analysis was conducted for basic propulsion thrust levels of 250K and 350K. The "Vehicle Applications Package, 250K Module" supplied by the Air Force for this study is reprinted in Appendix I. Appendix I also summarizes pertinent differences between the 250K package and the 350K package. The six application study vehicles along with the gross weights and ΔV requirements are shown in figures 1 and 2 for the 250K and 350K engines, respectively.

(U) Performance Index is the figure of merit used for evaluating installed engine performance and is defined as follows:

$$W_x = \left[\frac{W_g - 2A_i}{W_g}\right]\left\{W_{bc} - K\left[W_e + 1.5\,A_f + 0.02\left(W_g - W_{bc}\right)\left(1 + \frac{5}{r}\right)\left(\frac{500,000}{W_g - W_{bc}}\right)^{1/3}\right]\right\}$$

W_x = Performance Index

W_g = Stage gross weight (lb_m)

A_i = Interstage surface area (ft^2)

W_{bc} = Corrected stage burnout weight (lb_m)

K = Stage weight-growth constant

W_e = Engine installation weight, including modules, thrust structure, propellant lines, etc. (lb_m)

A_f = Fairing surface area (ft^2)

r = Oxidizer-to-fuel tank mixture ratio, by weight.

Performance Index represents stage burnout weight, less the weight of the engine, engine influenced hardware, and other stage inert weight.

(U) A computer program was used to make all performance index calculations. One main program was written for each case, to permit mission and installation requirements to be tailored for each application. To

perform the Performance Index calculation for a typical case, the following basic input data were required:

1. Mission and application definition
2. Installation weight and geometry
3. Engine performance and geometry.

1. Mission and Application Definition

(C) To perform the Performance Index calculation, stage gross weight (W_g) and stage weight-growth constant (K) were defined in the vehicle applications package for each case. (Refer to Appendix I.) The stage weight-growth constant (K) provides a means for differentiating the inert weight differences between expendable and recoverable upper and lower stage applications (i.e., the K value is greater than 1.0 for recoverable Cases 4, 5, and 6 only).

(U) The corrected stage burnout weight is determined from propellant consumption and is calculated from vehicle trajectory and engine performance data. The vehicle application package provided the following trajectory data for each application: (Refer to Appendix I for specific cases.)

1. Altitude versus velocity (Cases 1 through 6)
2. Trajectory loss parameter, L, versus velocity (Cases 1 through 6)
3. Stage burnout weight correction factor versus maximum fairing skirt diameter or height-to-body diameter (Cases 1, 3, 4, and 5).

(U) Propellant consumption for each application was calculated by using a modified form of the ideal velocity equation and integrating in a stepwise manner for velocity increments of 200 ft/sec throughout the entire trajectory. This specified equation and procedure are provided in Appendix I.

(U) The provided altitude versus velocity curve was used to determine the ambient pressure surrounding the engine and thus to determine engine module instantaneous delivered specific impulse for each velocity increment in the integration sequence. The trajectory loss parameter, L, data were used in curve fit form in the computer program to calculate the trajectory losses (gravity, drag, and steering) for each velocity increment during the integration process. Stage burnout weight correction factor is applied directly to the computed burnout weight and reflects the Performance Index penalties induced by skirt drag for cases where skirt diameter exceeds stage diameter.

2. Installation Weight and Geometry

(U) All weight associated with the engine installation is included in the engine installation weight term (W_e). The installation weight (W_e) and fairing (A_f) and interstage (A_i) area terms accent the weight and geometry

of engine installation and influenced hardware for each application in the Performance Index equation.

(U) By using the constraints and vehicle installation diagrams provided in the vehicle application package, installation weights for each application were determined for: (1) thrust structure, (2) propellant feedlines, (3) heatshield, (4) pressurization system, (5) thrust vector control systems, and (6) failure detection systems. These weights plus the engine module(s) weight comprise the (W_e) term shown in the Performance Index equation. A detailed description of the installation systems outlined above is provided in Section VI.

(U) Aerodynamic fairings (A_f) and interstage area (A_i) are directly associated with engine and stage geometry. For each case, the fairing area (A_f) and interstage area (A_i) terms in the performance index equation are calculated in the main body of the computer program.

3. Engine Performance

(U) Engine module performance data required to calculate the correct stage burnout weight for the Performance Index calculations are given in Appendix II. These data can be used for determining performance at given thrust levels and for the constant propellant flow rate engine module. Data are given for vacuum and all altitude conditions along with the optimum transition altitude for the two-position nozzle. The transition altitude is the point at which the nozzle is actuated from a retracted position to a fully extended position.

(U) Oxidizer-to-fuel tank mixture ratio, r, is an input item in the Performance Index equation and reflects the importance of propellant bulk density on stage inert weight (especially for the recoverable applications).

B. OTPIMIZATION TECHNIQUES

(U) The objective of the Applications Study was to select a basic engine module and to evaluate performance in all of the vehicle application cases. For the purpose of this applications study, the basic engine module is defined as developing 250K (or 350K) at vacuum conditions. This basic engine module defines an engine flow rate characteristic as a function of mixture ratio. The flow rate characteristic as well as such parameters as chamber pressure remain constant for all six vehicle cases. The selected basic engine module is referred to as the common module.

(U) In the final optimization process, the thrust of any module was limited to 250K (or 350K) even though the flow rate was held constant (i.e., it is possible to exceed the basic module thrust by increasing expansion ratio, but maintaining flow constant).

(U) A basic assumption was that all cases are equal in importance. Therefore, the selection of the basic or common module was based on performance analysis of all the cases; however, because of the wide range of propulsion requirements within the six application cases, the optimum nozzle expansion ratios and optimum tank mixture ratios vary widely. Further, these optimum values vary with the basic level of propellant flow rate, (i.e., common module area ratio). Further, as noted in Section VI, the weight of the various installation equipment varies with engine mounting location, which must also be optimized. The effect of varying bell nozzle contour must be evaluated; and, the primary area ratio (point at which the two-position nozzle translates) must be optimized.

(U) The following variables are involved in the optimization of each case:

1. Tank mixture ratio
2. Nozzle expansion ratio
3. Engine mounting location
4. Nozzle contour
5. Primary area ratio (lower stages only).

(U) Given a basic engine constraint such as constant thrust (propellant flow will vary with expansion ratio) or constant propellant flow rate (thrust will vary with expansion ratio), expansion ratio, mixture ratio and engine location can be optimized by the procedures as illustrated in figures 12 and 13. Figure 12 shows the upper stage technique and figure 13 the lower stage technique. Engine nozzle contour and primary area ratio optimization is accomplished by going through the initial optimization technique with different values of these parameters.

1. Upper Stage Optimization

(U) Figure 12 is a pictorial representation of the upper stage optimization procedure. Initially, the Performance Index of an upper stage case as a function of mixture ratio for various expansion ratios is plotted at a constant mount height. For each expansion ratio, the peak W_x determined the corresponding optimum mixture ratio. This procedure is then repeated for other mount heights. The peak values of W_x thus obtained are then plotted as a function of engine mount height. These curves are at the optimum mixture ratio for each expansion ratio. The final step in this optimization procedure is plotting the determined peak W_x values as a function of the case expansion ratio to obtain the complete optimization.

2. Lower Stage Optimization

(U) The optimization procedure for the lower stage cases is the same as for the upper stage cases except the procedure includes the additional variable of engine mount radius. A pictorial representation of the lower stage optimization procedure is presented in figure 13.

(U) During initial studies the optimum engine mount height and gimbal point radius were determined by the procedure outlined above. Based on the experience gained for each case, the computer was thereafter used directly to determine optimum engine mount height and gimbal point radius for succeeding Performance Index calculations. This was accomplished by "boxing in" the optimum engine mounting location with a parametric range of mount height and gimbal point radii locations for each expansion ratio point. The optimum mount location for each point was then read directly from the computer output. This permits the initial W_x vs mixture ratio curves in the upper and lower stage optimization (figures 12 and 13) to be prepared for the optimum mounting point.

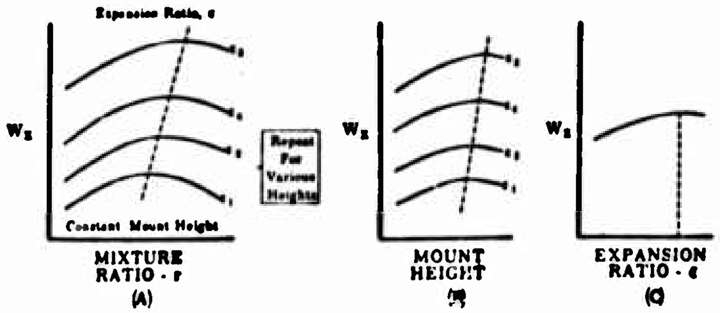


Figure 12. Optimization Procedure for an Upper Stage Case FD 21998A

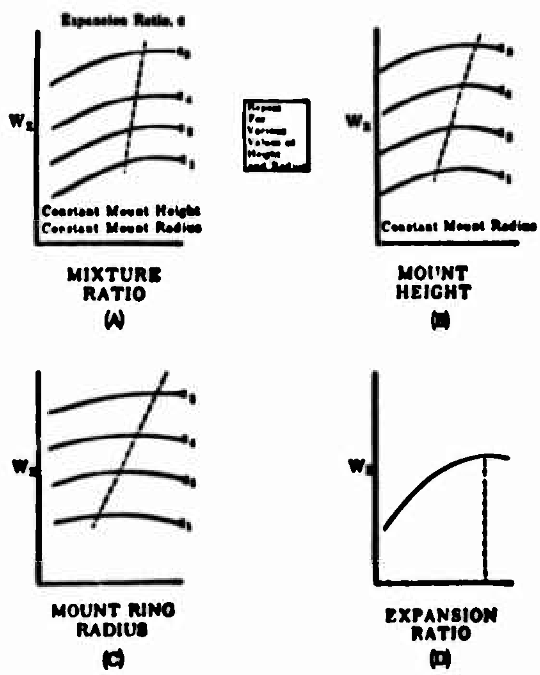


Figure 13. Optimization Procedure for Lower Stage Case FD 21999A

3. Engine Module Size Selection

(U) The basic engine module combustion chamber-turbopump power package size was selected to produce the highest summed value of Performance Index for all six vehicle cases. During the study, it was found that Performance Index increased with module size in all six cases but the effect was especially strong in the lower stages. The module size was also selected so that no stage exceeded the specified vacuum thrust level (250K or 350K). Upper stages reach this limit first as they tend to optimize with high area ratio nozzles (thrust increases with engine expansion ratio for a constant module size). The size selection optimization technique must, thus, balance these two opposing upper and lower stage requirements. The technique used for selecting the size of the module (common engine module) for the final W_x values in this report was based on a summation of the performance of all six cases on an equal basis. This was done for a series of basic module sizes and the module size giving the best level of summed performance selected. No engine was allowed to exceed 250K vacuum thrust in this process. Other techniques for determining the common module size are discussed in Section V.

(U) The technique used for selecting the common module size is shown pictorially in figure 14. The 250K engine module size is used as the example in this discussion, however, the same technique was used for the 350K engine.

(C) As the initial step in the common module selection, four basic engine sizes were selected:

1. Engine power package sized to deliver 250K vacuum thrust at an area ratio of 100; propellant flow at mixture ratio of 6 = 549.9 lb/sec.

2. Engine power package sized to deliver 250K vacuum thrust at an area ratio of 200; propellant flow at mixture ratio of 6 = 539.3 lb/sec.

3. Engine power package sized to deliver 250K vacuum thrust at an area ratio of 300; propellant flow at mixture ratio of 6 = 534 lb/sec.

4. Engine power package sized to deliver 250K vacuum thrust at an area ratio of 400; propellant flow at mixture ratio of 6 = 531.0 lb/sec.

(U) Each of these engines, thus, has a different sized power package that delivers 250K vacuum thrust at only the one engine expansion ratio. These engine sizes are identified in figure 14 as "Engine Module Size - ϵ_M."

(U) During initial studies the optimum engine mount height and gimbal point radius were determined by the procedure outlined above. Based on the experience gained for each case, the computer was thereafter used directly to determine optimum engine mount height and gimbal point radius for succeeding Performance Index calculations. This was accomplished by "boxing in" the optimum engine mounting location with a parametric range of mount height and gimbal point radii locations for each expansion ratio point. The optimum mount location for each point was then read directly from the computer output. This permits the initial W_x vs mixture ratio curves in the upper and lower stage optimization (figures 12 and 13) to be prepared for the optimum mounting point.

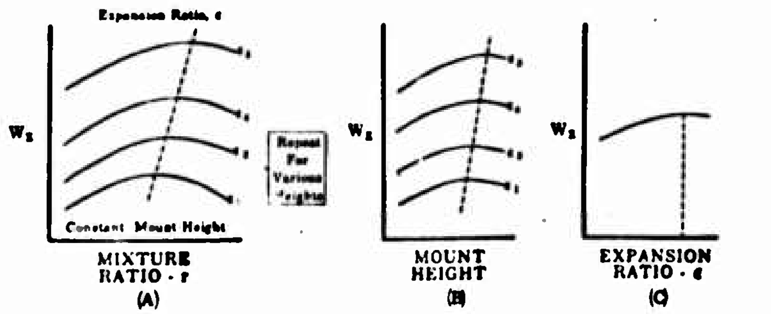


Figure 12. Optimization Procedure for an Upper Stage Case FD 21998A

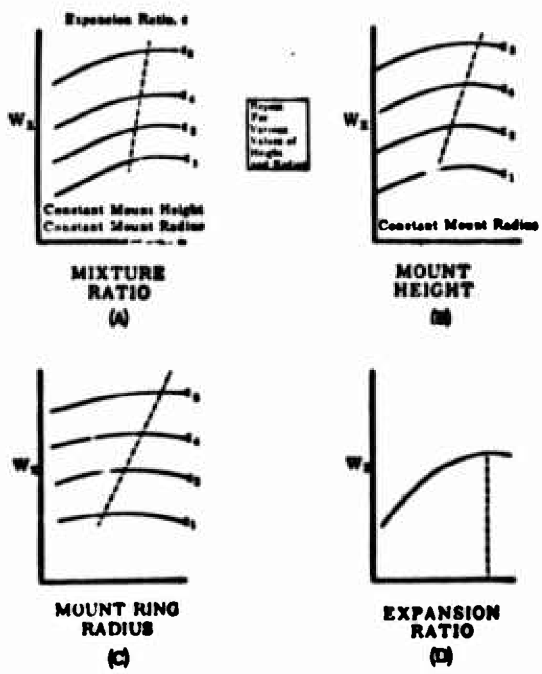


Figure 13. Optimization Procedure for Lower Stage Case FD 21999A

3. Engine Module Size Selection

(U) The basic engine module combustion chamber-turbopump power package size was selected to produce the highest summed value of Performance Index for all six vehicle cases. During the study, it was found that Performance Index increased with module size in all six cases but the effect was especially strong in the lower stages. The module size was also selected so that no stage exceeded the specified vacuum thrust level (250K or 350K). Upper stages reach this limit first as they tend to optimize with high area ratio nozzles (thrust increases with engine expansion ratio for a constant module size). The size selection optimization technique must, thus, balance these two opposing upper and lower stage requirements. The technique used for selecting the size of the module (common engine module) for the final W_x values in this report was based on a summation of the performance of all six cases on an equal basis. This was done for a series of basic module sizes and the module size giving the best level of summed performance selected. No engine was allowed to exceed 250K vacuum thrust in this process. Other techniques for determining the common module size are discussed in Section V.

(U) The technique used for selecting the common module size is shown pictorially in figure 14. The 250K engine module size is used as the example in this discussion, however, the same technique was used for the 350K engine.

(C) As the initial step in the common module selection, four basic engine sizes were selected:

1. Engine power package sized to deliver 250K vacuum thrust at an area ratio of 100; propellant flow at mixture ratio of 6 = 549.9 lb/sec.

2. Engine power package sized to deliver 250K vacuum thrust at an area ratio of 200; propellant flow at mixture ratio of 6 = 539.3 lb/sec.

3. Engine power package sized to deliver 250K vacuum thrust at an area ratio of 300; propellant flow at mixture ratio of 6 = 534 lb/sec.

4. Engine power package sized to deliver 250K vacuum thrust at an area ratio of 400; propellant flow at mixture ratio of 6 = 531.0 lb/sec.

(U) Each of these engines, thus, has a different sized power package that delivers 250K vacuum thrust at only the one engine expansion ratio. These engine sizes are identified in figure 14 as "Engine Module Size - ϵ_M."

(C) In the next step, each of the six vehicle cases is optimized for the four different sized engines, i.e., ϵ_M = 100, 200, 300, and 400. As shown in figure 14, the initial working curves are a plot of W_x vs mixture ratio with lines of constant engine expansion ratio; each plot accomplished for one engine size (ϵ_M). These are then cross plotted vs engine expansion ratio with all four engine sizes (ϵ_M) on one plot. Peak values of W_x for each engine size are next plotted vs engine module size (ϵ_M). All six vehicle cases are then plotted on one curve as percentages of peak W_x and then summed and replotted as one curve. The point at which this summed W_x curve peaks defined the best basic expansion ratio or engine module size for all six vehicles.

(C) Once the common engine size (basic expansion ratio) had been selected, it was possible to examine the performance of each vehicle case to determine its optimum performance with this power package. Cases 2 and 6 tend to optimize at expansion ratios of 400 or higher; therefore, because there is a limit of 250K vacuum thrust, these engines could not exceed the 250 expansion ratio selected for the common module size. All other cases optimized at some expansion ratio lower than the basic expansion ratio.

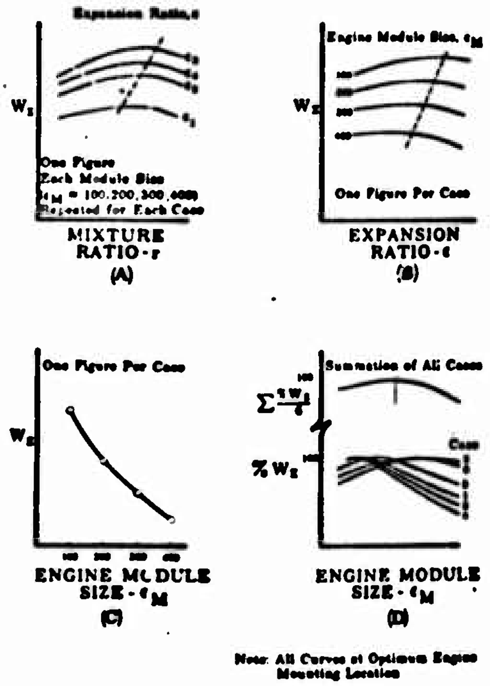


Figure 14. Optimization Procedure for Engine Module Size   FD 22000A

(U) The effects of nozzle contour are not directly within the case optimization procedure. This is also true for the primary nozzle area ratio (two-position nozzle engines) for the lower stages. Optimization of the primary nozzle area ratio is not necessary for the upper stages because primary nozzle area ratio is selected only to give the minimum stowed length for the engine. Primary nozzle area ratio does not affect engine performance because upper stage engines would operate only with the nozzle extended. The complete optimization procedure is repeated for various nozzle contours for all cases and with various primary area ratios in the lower stage cases.

C. COMMON MODULE SELECTION (250K MODULE)

1. Case Optimization

(C) The procedure outlined in the previous paragraphs was used to optimize each application case and to select a common module engine. The common module chosen develops 250K vacuum thrust at a nozzle expansion ratio of 250 with a minimum surface area bell nozzle contour.

(U) The evaluation and optimization of the upper stages for selection of the common module was conducted using a minimum surface area bell nozzle. Lower stages were based on maximum performance contours. Use of these contours was based on contour evaluation studies discussed later in this section. The basic curves used are shown only for optimum or selected engine locations, because the described computer technique was used for these final studies, and also for clarity of presentation.

(C) The matrix of mixture ratio and expansion ratio that establish the optimum case expansion ratio were completed for module engine sizes 100, 200, 300, and 400. By definition, these various engine sizes will develop 250K vacuum thrust at a nozzle expansion ratio of 100, 200, 300, and 400. The following paragraphs present the working and final curves used for each vehicle case to make the common module selection.

a. Case 1 - Expendable Lower Stage

(C) For the format used in the presentation of the Case 1 curves, refer to Paragraph B, Engine Module Size selection. Figures 15 through 18 show Performance Index of Case 1 as a function of mixture ratio and expansion ratio for engine module sizes (ϵ_M) of 100, 200, 300, and 400, respectively. These figures are for the optimum engine mount location. Figure 19 is the cross-plot of figures 15 through 18 and shows a Performance Index line of optimum expansion ratios for each module size (ϵ_M). Figure 20 is the cross-plot of the curve peaks on figure 19 as a function of engine module size (ϵ_M). Expansion ratio, mixture ratio, and mount location are, thus, optimum for each point on figure 20.

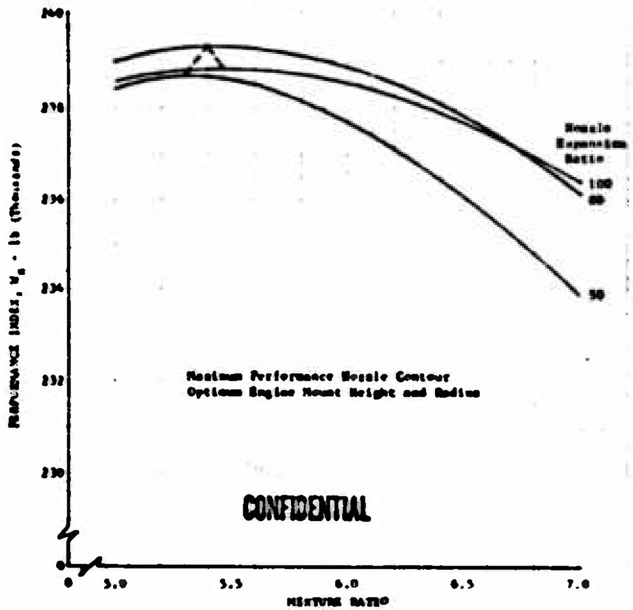


Figure 15. Performance Index vs Mixture Ratio for Case 1, 250K Module (ϵ_M = 100) DF 57527

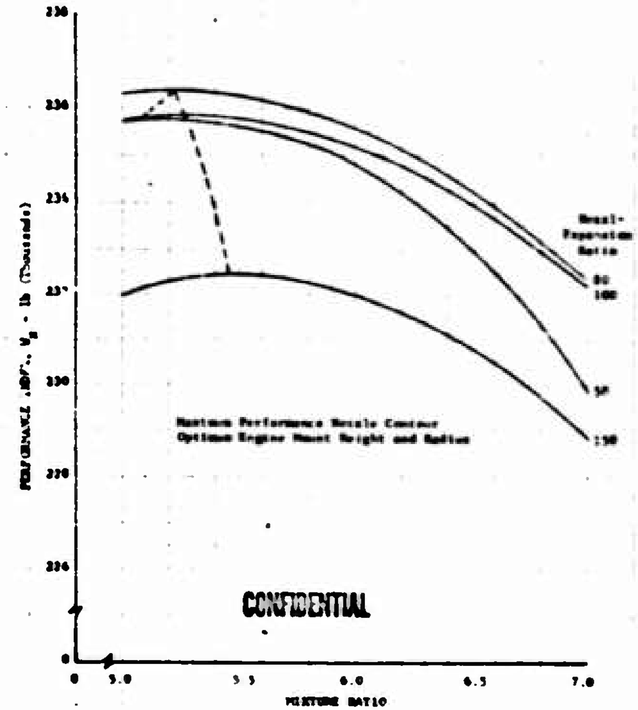


Figure 16. Performance Index vs Mixture Ratio for Case 1, 250K Module (ϵ_M = 200) DF 57528

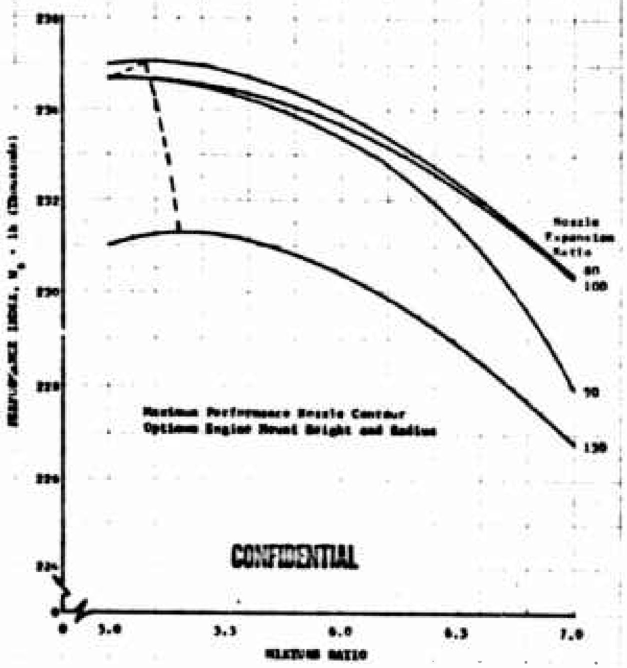


Figure 17. Performance Index vs Mixture Ratio for Case 1, 250K Module (ϵ_M = 300) DF 57529

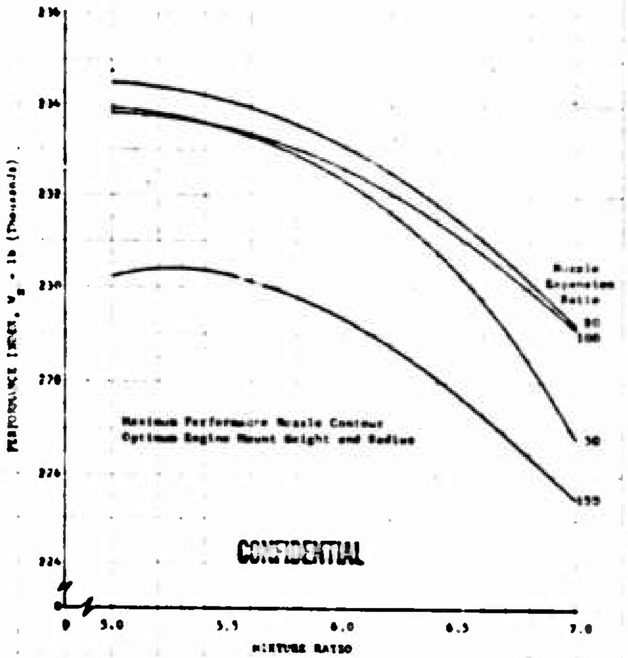


Figure 18. Performance Index vs Mixture Ratio for Case 1, 250K Module (ϵ_M = 400) DF 57530

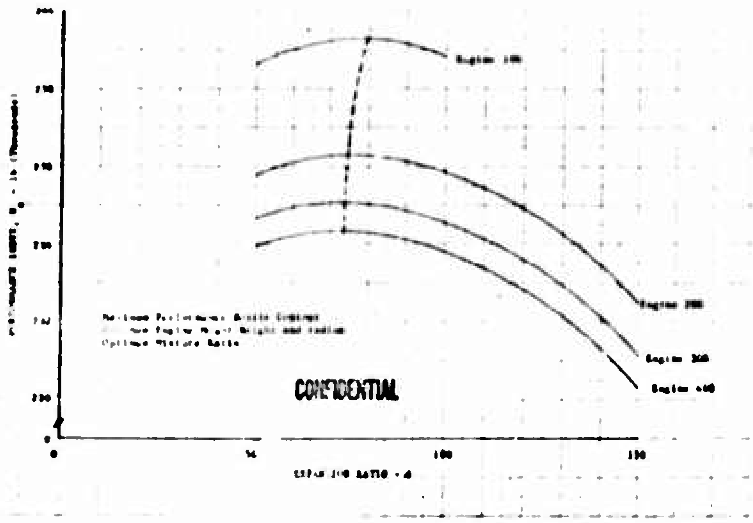


Figure 19. Performance Index vs Expansion Ratio for Case 1, 250K Module DF 57531

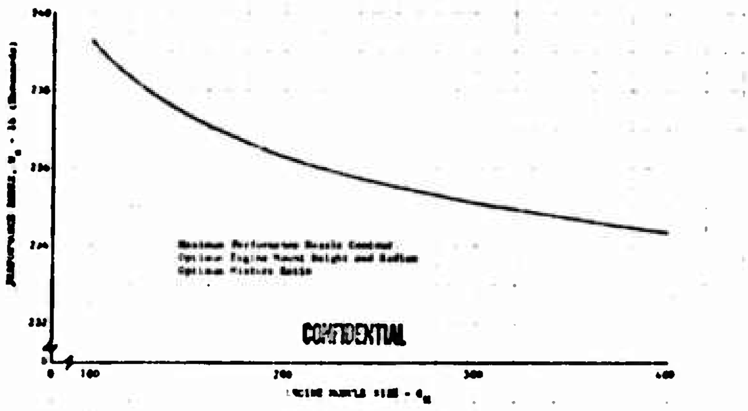


Figure 20. Performance Index vs Engine Module Size for Case 1, 250K Module DF 57532

b. Case 2 - Expendable Upper Stage

(C) The engine module size selection technique has been condensed for Cases 2, 5, and 6. The engine module size ranges considered for both Cases 2 and 6 produce highest performance at an expansion ratio of 400 (or higher). However, with an engine module sized to produce 250K at an expansion ratio of 100, it will produce more than 250K at expansion ratios above 100. Because one of the study ground rules is that thrust cannot exceed 250K, it is not possible to optimize expansion ratio at ratios higher than the ratio used for module size. This means that instead of having a series of expansion ratio lines for a given module size (ϵ_M) on the first series of curves of Performance Index versus mixture ratio (as outlined in the Engine Module Size Selection procedure), there would be only one line of interest. This line would represent the maximum performance attainable with this module size. Instead of making four separate figures as for the lower stage cases, all of the module sizes, ϵ_M, are plotted on one curve of Performance Index versus mixture ratio, figure 21. Further, additional engine module sizes of 80 and 150 were added to the 100, 200, 300, and 400 sizes used in the lower stages to aid in preparation of the cross-plots. The second series of curves (in the module size selection procedure) can be eliminated and the cross-plot of Performance Index versus engine module size (ϵ_M) made from the first curve, figure 21. This cross-plot is shown as figure 22 for Case 2.

c. Case 3 - Expendable Single Stage-to-Orbit

(C) Figures 23 through 26 show Performance Index of Case 3 as a function of mixture ratio and expansion ratio for basic engine module sizes (ϵ_M) of 100, 200, 300, and 400. Figure 27 is a cross-plot of figures 23 through 26 and shows the Performance Index line of optimum expansion ratios for each engine module size (ϵ_M). Figure 28 is a cross-plot of the curve peaks on figure 27 as a function of engine module size (ϵ_M).

d. Case 4 - Recoverable Lower Stage

(C) Figures 29 through 32 show Performance Index of Case 4 as a function of mixture ratio and expansion ratio for basic module sizes (ϵ_M) of 100, 200, 300, and 400. Figure 33 is a cross-plot of figures 29 through 32 and shows a Performance Index line of optimum expansion ratios for each engine module size (ϵ_M). Figure 34 is a cross-plot of the curve peaks on figure 33, as a function of engine module size (ϵ_M).

e. Case 5 - Recoverable Upper Stage, Pick-a-Back

(C) The data on Case 5 are presented similar to Case 2, however, the peak performance for Case 5 occurs at an engine module size of approximately 200. Therefore, module engine sizes of 200 or less are shown on the mixture ratio plot given in figure 35. Data for engine module sizes (ϵ_M) of 300 and 400 are shown in figures 36 and 37, respectively. The peaks of these three figures are shown on figure 38, which gives the optimum expansion ratio for the various engine module sizes. The data for engine module sizes less than 200 are plotted from figure 35. The final summary performance as a function of engine size is shown on figure 39.

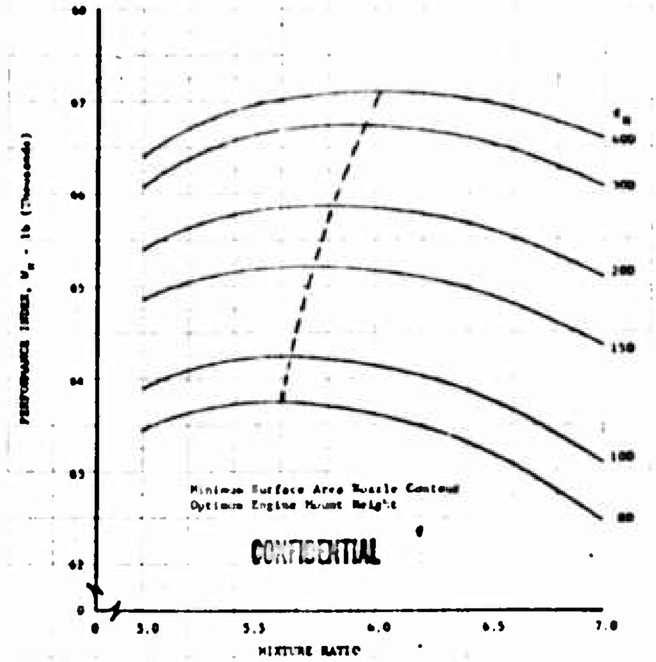


Figure 21. Performance Index vs Mixture Ratio for Case 2, 250K Module DF 57533

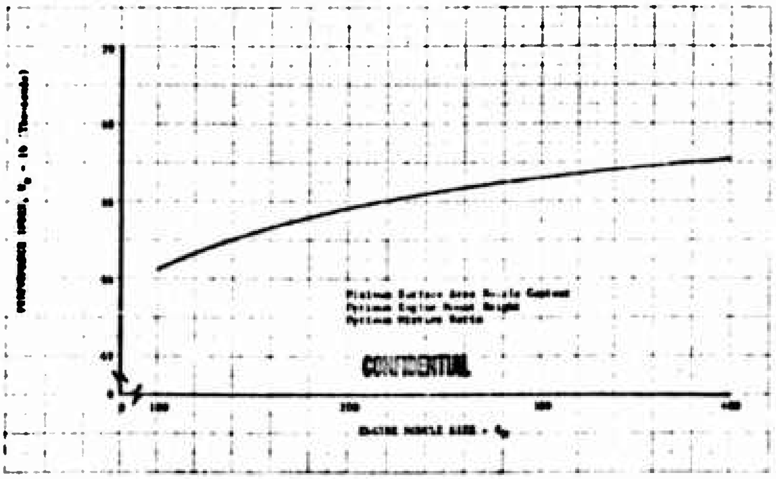


Figure 22. Performance Index vs Engine Module Size for Case 2, 250K Module DF 57534

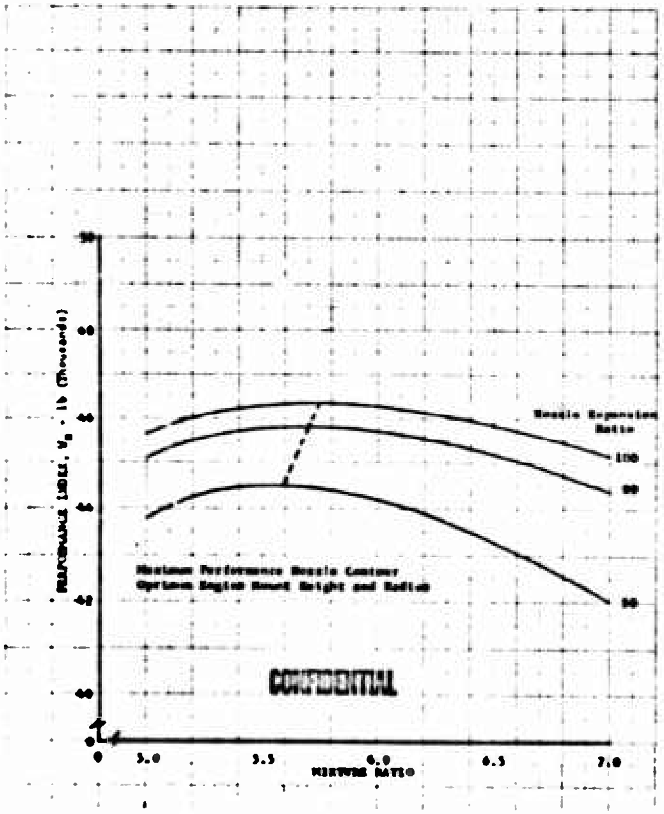


Figure 23. Performance Index vs Mixture Ratio for Case 3, 250K Module (ϵ_M = 100) DF 57558

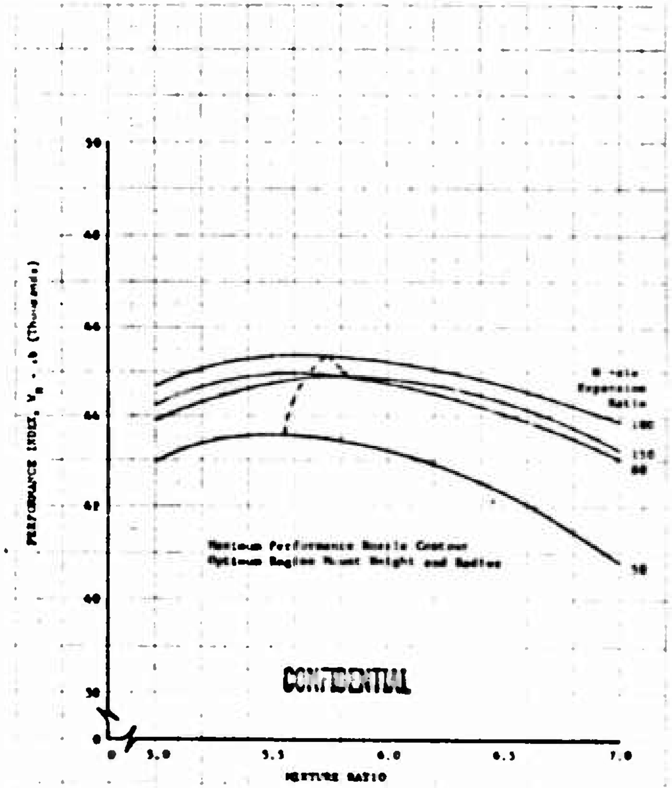


Figure 24. Performance Index vs Mixture Ratio for Case 3, 250K Module (ϵ_M = 200) DF 57535

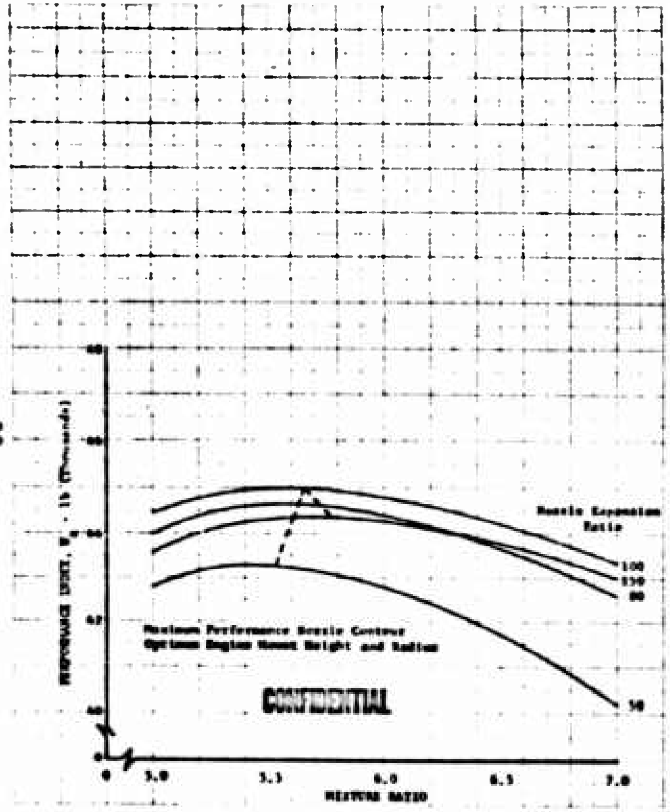


Figure 25. Performance Index vs Mixture Ratio for Case 3, 250K Module (ϵ_M = 300) DF 57536

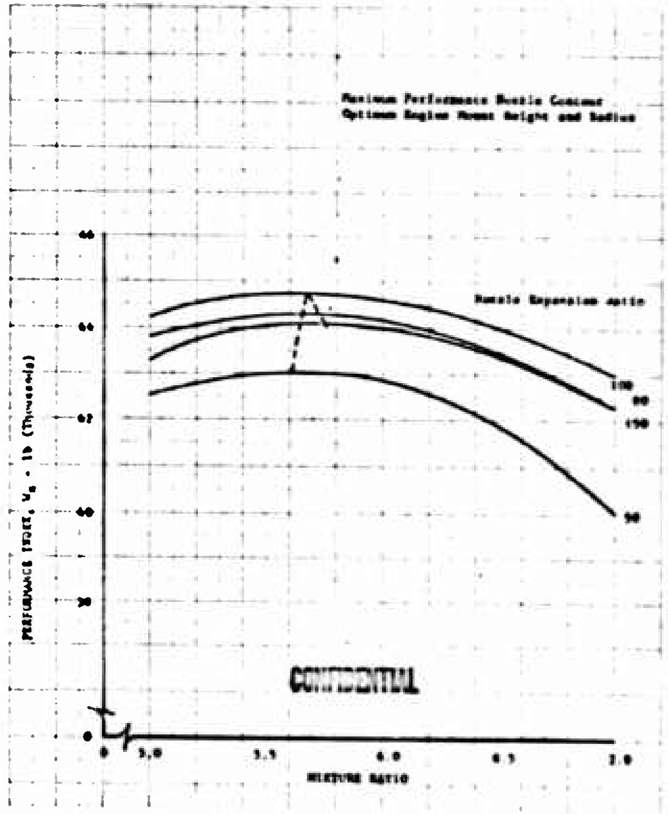


Figure 26. Performance Index vs Mixture Ratio for Case 3, 250K Module (ϵ_M = 400) DF 57537

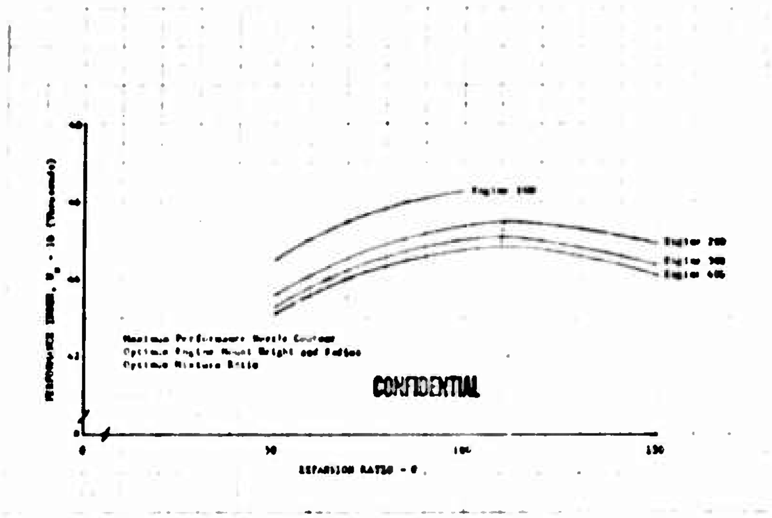


Figure 27. Performance Index vs Expansion Ratio for Case 3, 250K Module DF 57538

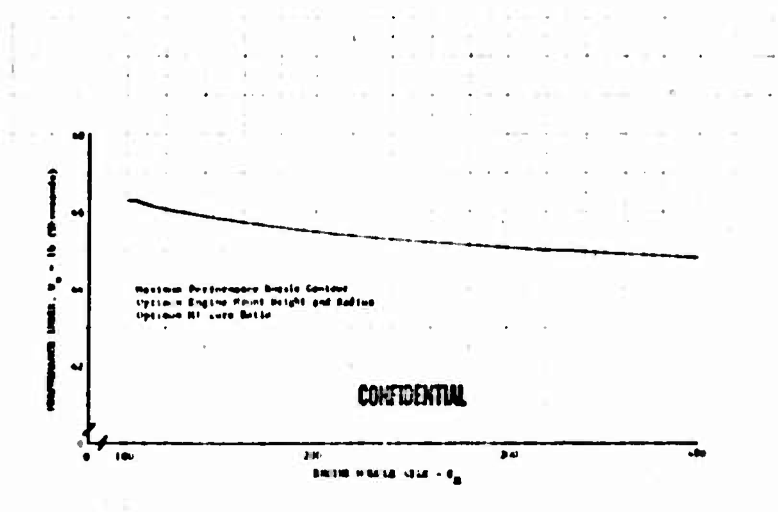


Figure 28. Performance Index vs Engine Module Size for Case 3, 250K Module DF 57539

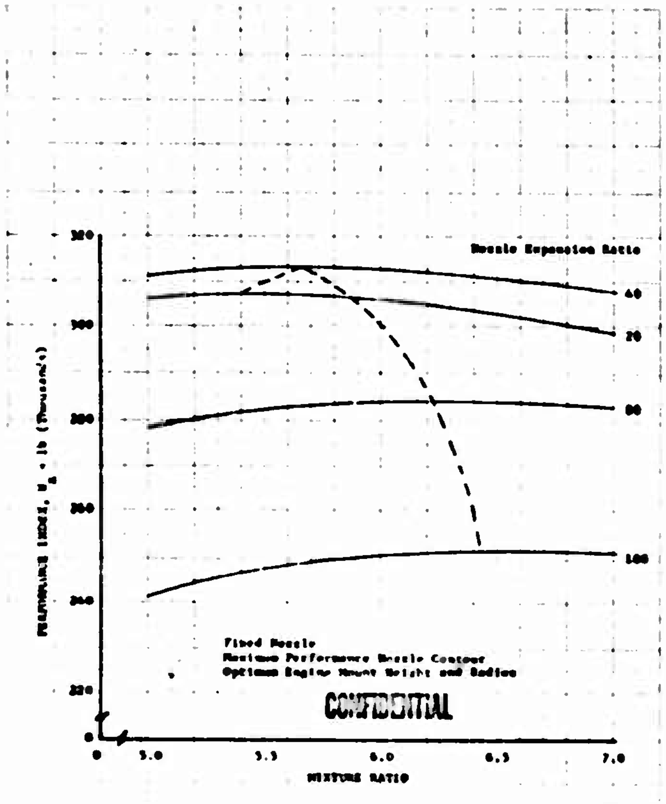


Figure 29. Performance Index vs Mixture Ratio for Case 4, 250K Module (ϵ_M = 100) DF 57540

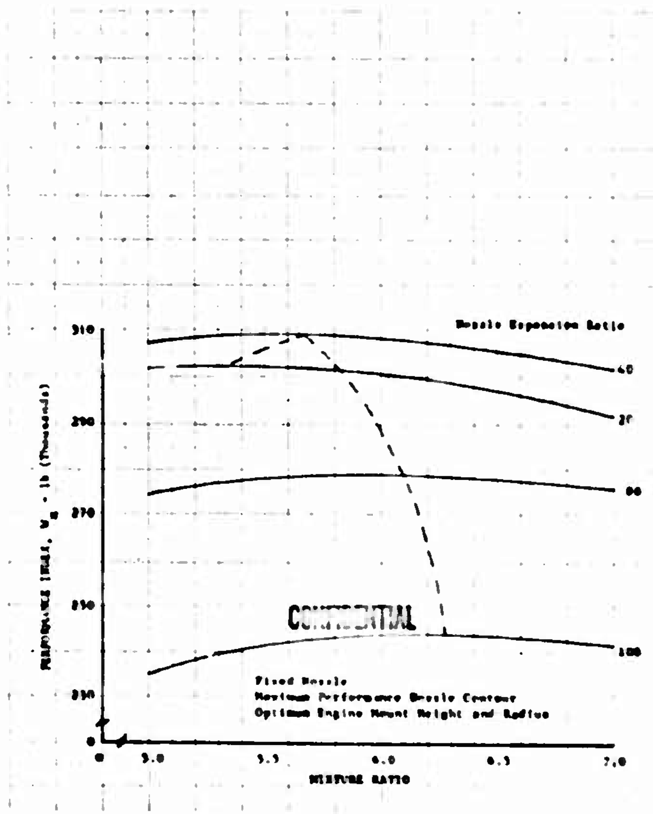


Figure 30. Performance Index vs Mixture Ratio for Case 4, 250K Module (ϵ_M = 200) DF 57541

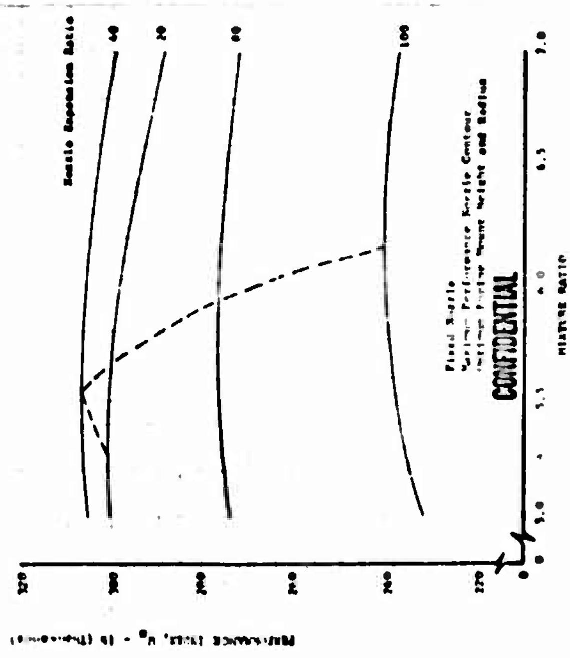


Figure 31. Performance Index vs Mixture Ratio for Case 4, 250K Module (ϵ_M = 300) DF 57542

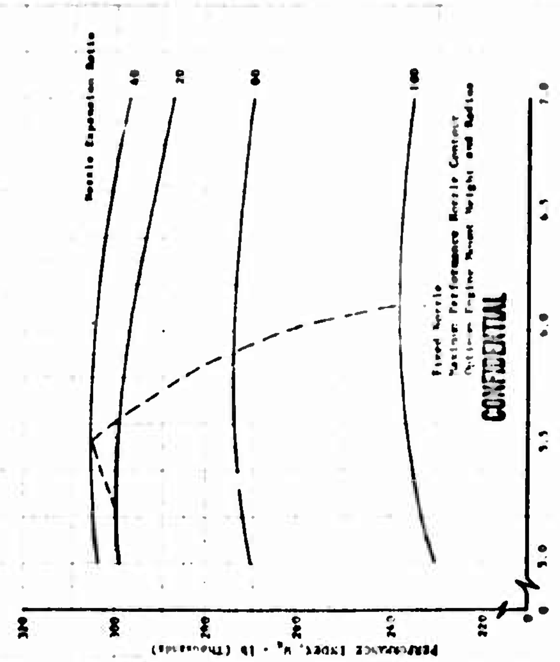


Figure 32. Performance Index vs Mixture Ratio for Case 4, 250K Module (ϵ_M = 400) DF 57543

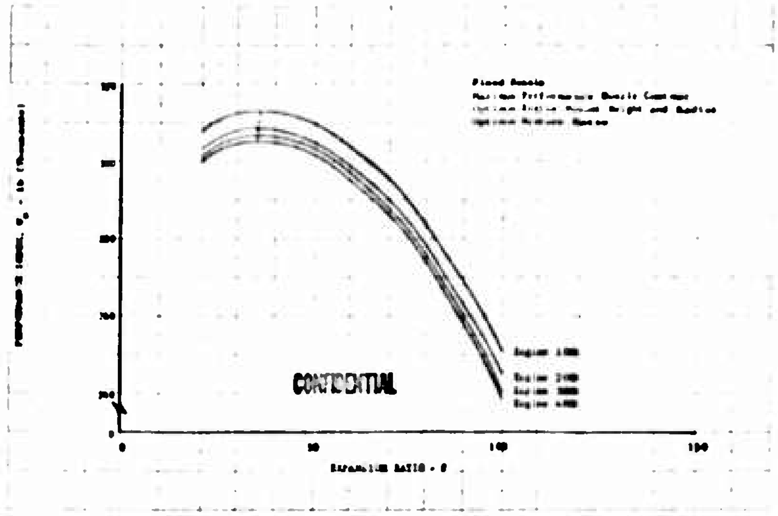


Figure 33. Performance Index vs Expansion Ratio for Case 4, 250K Module DF 57544

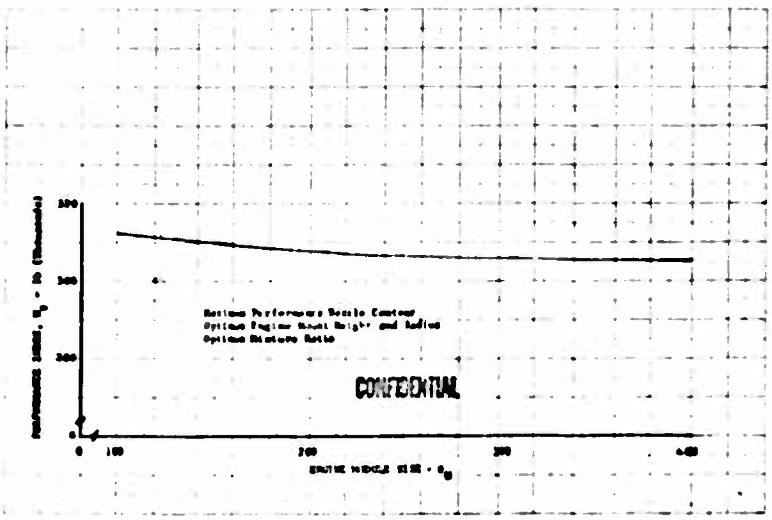


Figure 34. Performance Index vs Engine Module Size for Case 4, 250K Module DF 57545

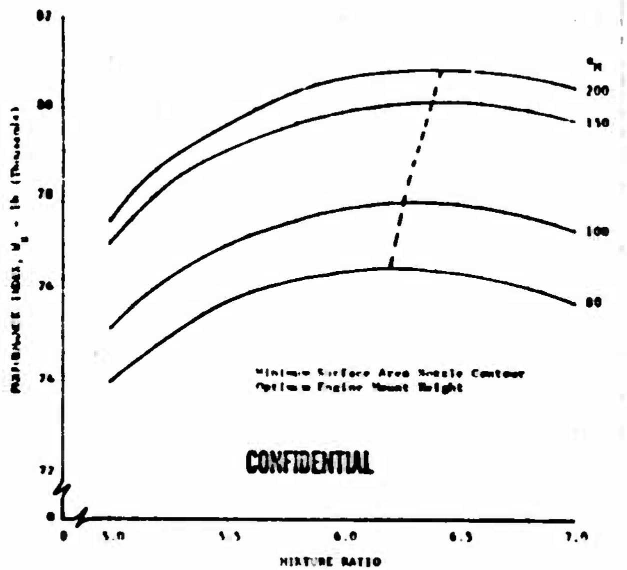


Figure 35. Performance Index vs Mixture Ratio for Case 5, 250K Module DF 57559

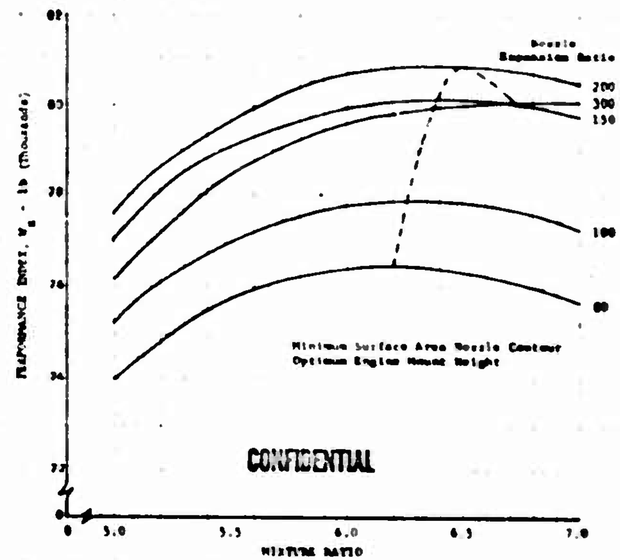


Figure 36. Performance Index vs Mixture Ratio for Case 5, 250K Module (ϵ_M = 300) DF 57560

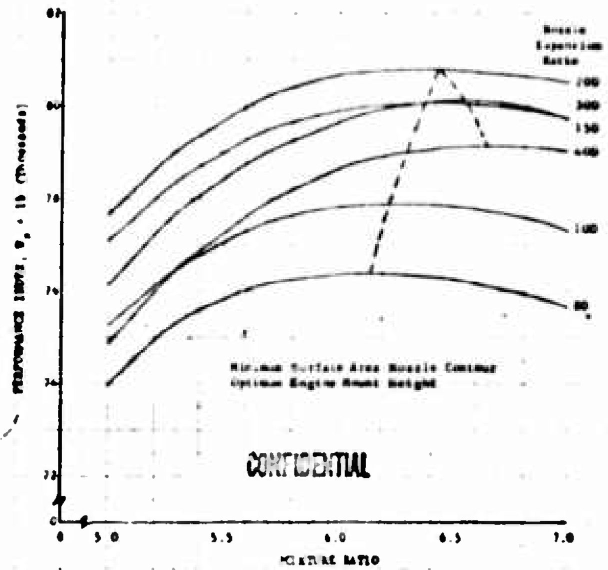


Figure 37. Performance Index vs Mixture Ratio for Case 5, 250K Module (ϵ_M = 400) DF 57561

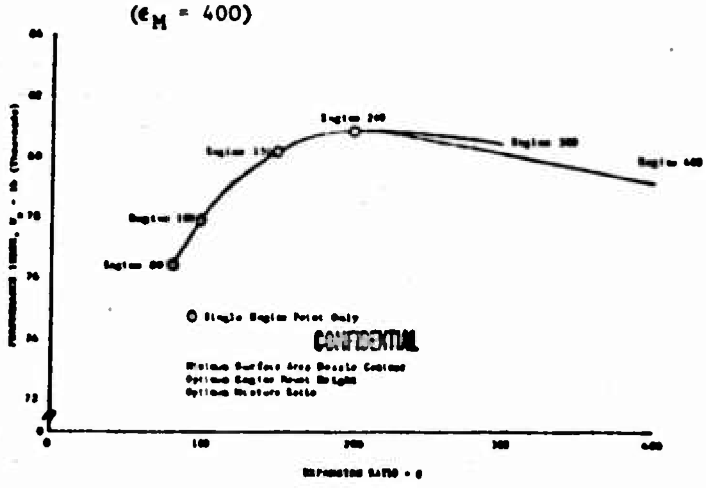


Figure 38. Performance Index vs Expansion Ratio for Case 5, 250K Module DF 57562

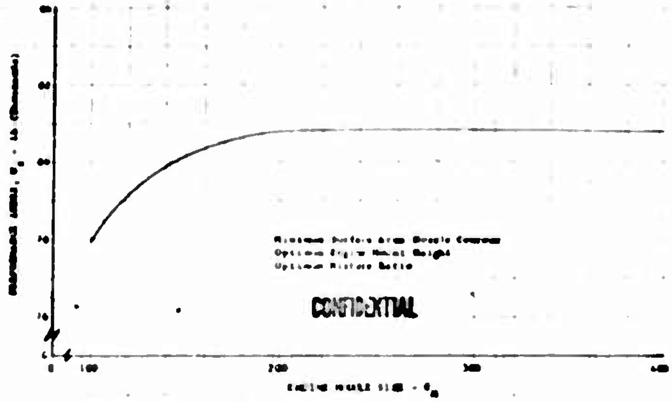


Figure 39. Performance Index vs Engine Module Size for Case 5, 250K Module

DF 57563

f. Case 6 - Recoverable Upper Stage, Tandem

(C) The data on Case 6 are presented in the same manner as Case 2. Figure 40 shows Performance Index of Case 6 as a function of mixture ratio and expansion ratio for various basic module sizes (ϵ_M). Figure 41 is a cross-plot of figure 40 and shows Performance Index as a function of module size.

2. Common Module Size

(U) The Performance Index versus engine module size (ϵ_M) curves for Cases 1 through 6 were converted into a percentage of the maximum Performance Index for each case and is shown on the lower portion of figure 42 for Cases 1 through 6. These curves were generated from the figures listed below:

Case 1	Figure 20
Case 2	Figure 22
Case 3	Figure 28
Case 4	Figure 34
Case 5	Figure 39
Case 6	Figure 41

(U) The six Performance Index curves (in percent) were then summed and divided by six and an average curve for all six cases presented at the top of figure 42. All points on figure 42 reflect optimum mixture ratio, case expansion ratio, and engine mount location.

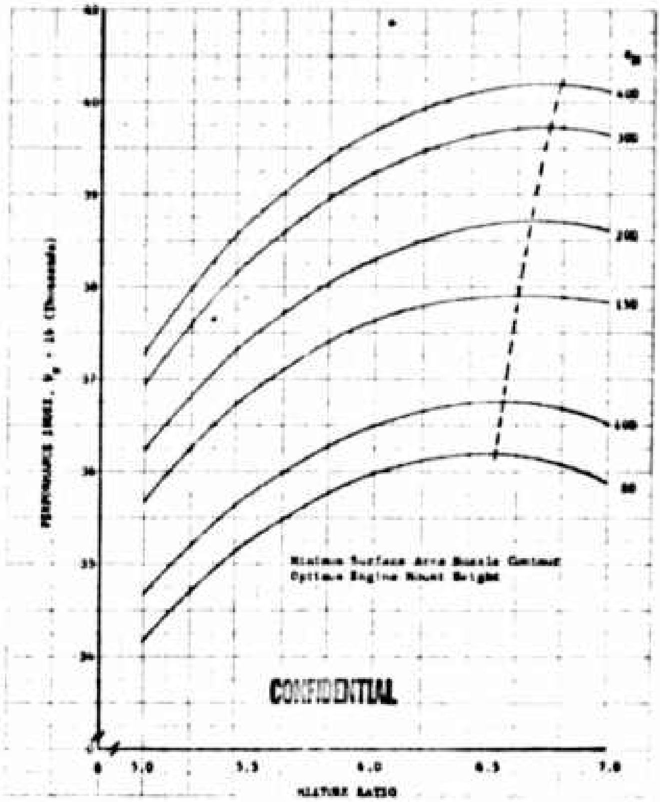


Figure 40. Performance Index vs Mixture Ratio for Case 6, 250K Module DF 57546

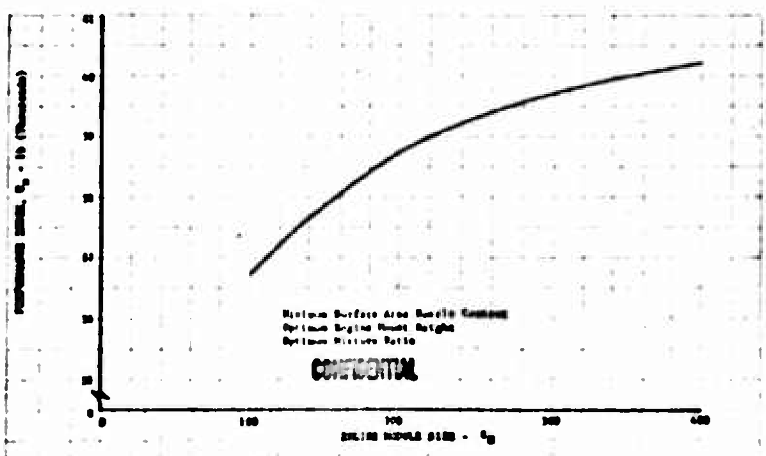


Figure 41. Performance Index vs Engine Module Size for Case 6, 250K Module DF 57547

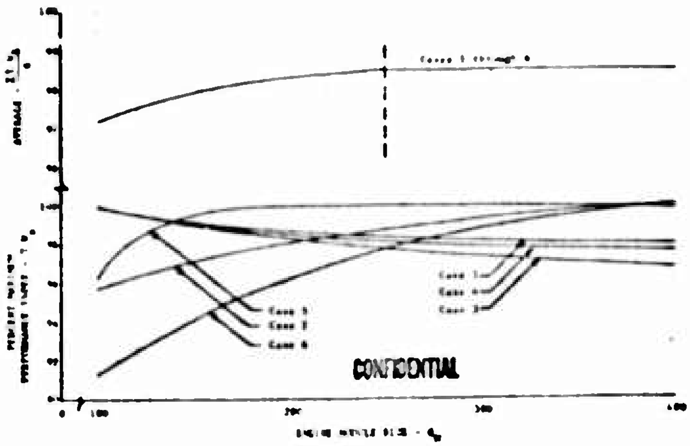


Figure 42. Percent Performance Index vs Engine Module Size, 250K Module DF 57571

(C) On figure 42, it is shown that the optimum engine module size is a power package that delivers 250K with an expansion ratio of 250.

(U) Figure 42 indicates that the lower stage applications (Cases 1, 3, and 4) are thrust sensitive and realize high Performance Indexes for low compromise expansion ratio and high flow rate. The upper stage applications are vacuum specific impulse sensitive and realize high Performance Indexes for high expansion ratios. The common engine module selection technique used makes the best performance compromise between the various cases.

3. Case Performance

(U) The optimum performance of each case was established using the selected common engine module and common engine module flow rate. These optimization procedures are the same as outlined previously (figures 12, 13, and 14) except calculations are made for the one established engine module size. The performance thus obtained is shown in table I.

a. Case 1 - Expendable Lower Stage

(C) The performance is shown in figure 43 as a function of mixture ratio and expansion ratio for the flow rate of the common module. The optimum expansion ratio was determined to be 75 as can be seen from figure 44, which is the cross-plot of figure 43. The optimum mixture ratio is 5.15 at an expansion ratio of 75.

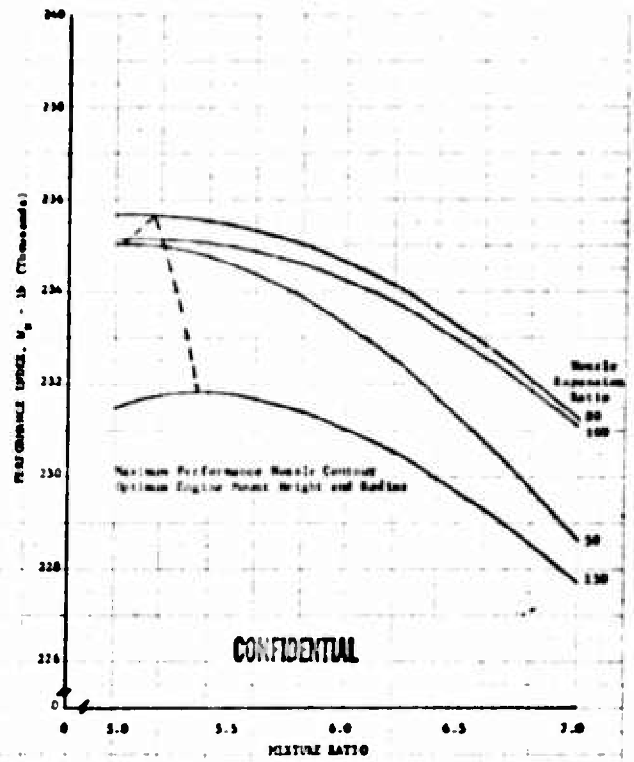


Figure 43. Performance Index vs Mixture Ratio for Case 1, 250K Module (ϵ_M = 250) DF 57548

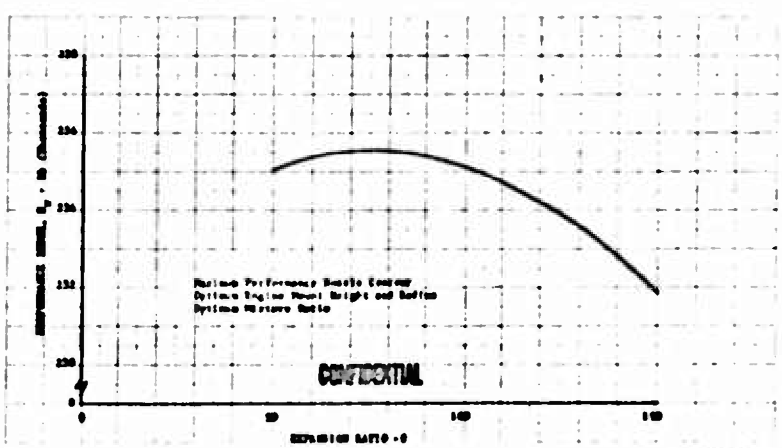


Figure 44. Performance Index vs Expansion Ratio for Case 1, 250K Module (ϵ_M = 250) DF 57551

b. Case 2 - Expendable Upper Stage

(C) The performance of Case 2 is shown in figure 45 as a function of expansion ratio and mixture ratio for a module size of 250. This application would tend to optimize at a higher expansion ratio due to a lack of strong length constraint. However, as noted in the Case 2 discussion, it is not possible to optimize the expansion ratio at ratios higher than that used for the common module because thrust would exceed 250K. The expansion ratio optimization curve is given in figure 46 at an expansion ratio of 250.

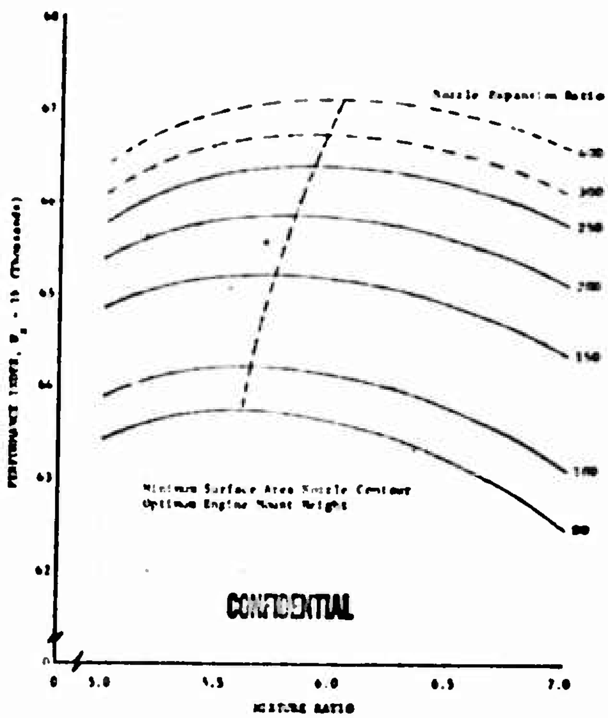


Figure 45. Performance Index vs Mixture Ratio for Case 2, 250K Module (ϵ_M = 250) DF 57572

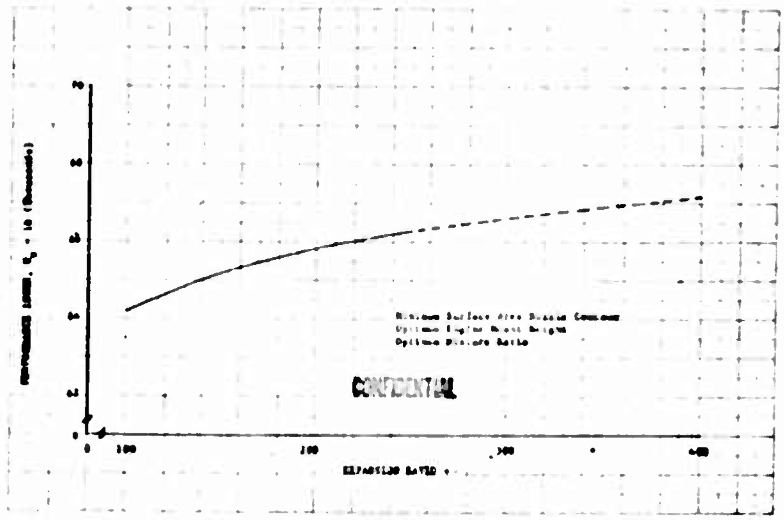


Figure 46. Performance Index vs Expansion Ratio for Case 2, 250K Module (ϵ_M = 250) DF 57573

c. Case 3 - Expendable Single Stage-to-Orbit

(C) Performance as a function of mixture ratio and expansion ratio is shown in figure 47 for the common engine module. Figure 48 gives the Performance Index of Case 3 as a function of expansion ratio for the common engine module. Figure 48 shows that the optimum expansion ratio for Case 3 is 110. Because Case 3 represents a single stage-to-orbit mission, both sea level and vacuum specific impulse are significant. Accordingly, a relatively high secondary expansion ratio (compared to Case 1) was required in Case 3 to provide the necessary vacuum performance needed for the latter portion of the mission. Optimum mixture ratio is 5.76 and relative to Case 1, shows the influence of increasing expansion ratio and the propellant bulk density sensitivity of this stage.

d. Case 4 - Recoverable Lower Stage

(C) Performance as a function of mixture ratio and expansion ratio is presented in figure 49. The optimum expansion ratio for the common engine module is shown in figure 50. Because of the low area ratio requirements and strong diameter constraints of this application, a fixed-position nozzle at an area ratio of 35 gives improved performance. The optimum mixture ratio is 5.52 at an expansion ratio of 35.

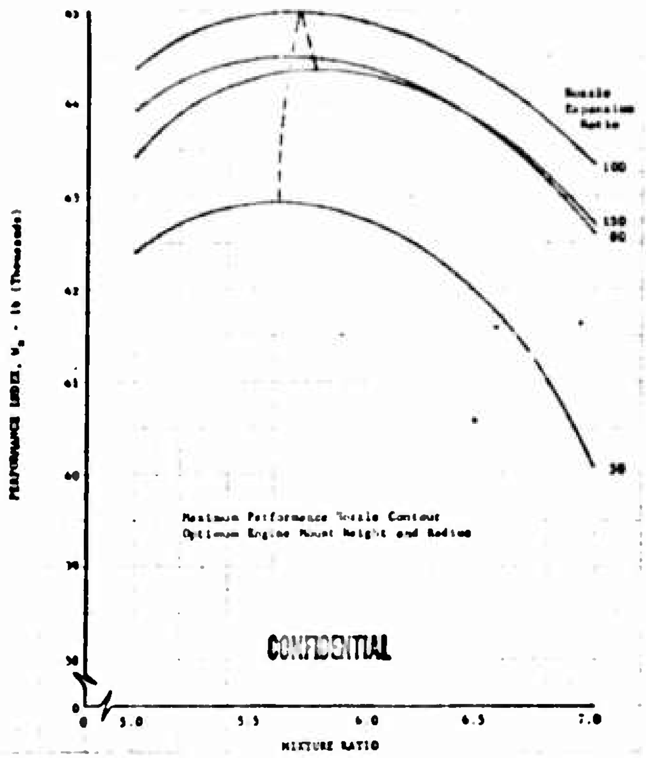


Figure 47. Performance Index vs Mixture Ratio for Case 3, 250K Module (ϵ_M = 250) DF 57550

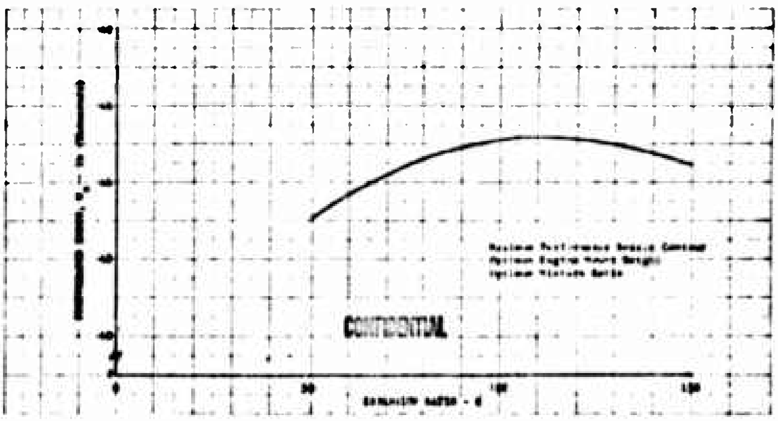


Figure 48. Performance Index vs Expansion Ratio for Case 3, 250K Module (ϵ_M = 250) DF 57553

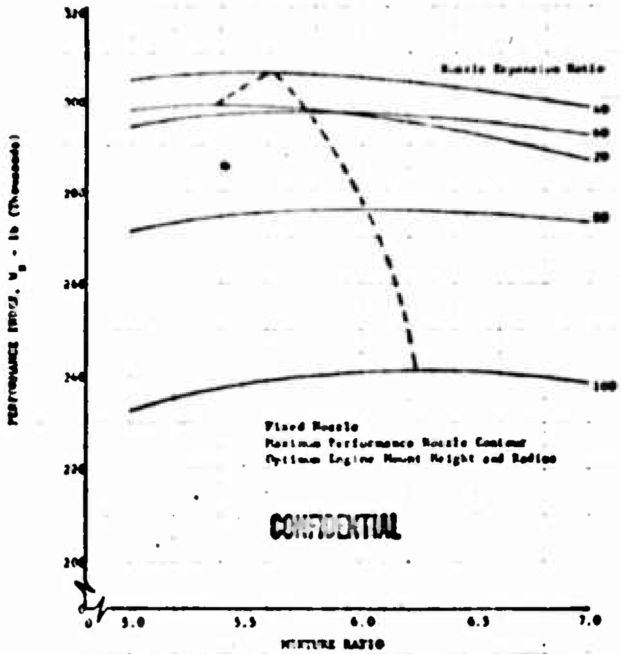


Figure 49. Performance Index vs Mixture Ratio for Case 4, 250K Module (ϵ_M = 250) DF 57549

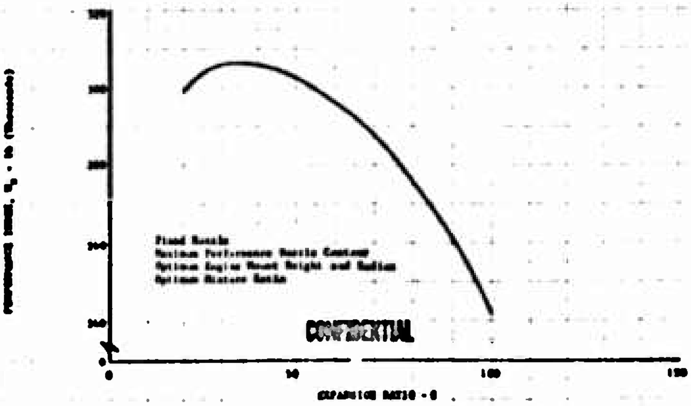


Figure 50. Performance Index vs Expansion Ratio for Case 4, 250K Module (ϵ_M = 250) DF 57552

e. Case 5 - Recoverable Upper Stage, Pick-a-Back

(C) Figure 51 shows the Performance Index for Case 5 as a function of mixture ratio and expansion ratio for the common engine module size. The optimum expansion ratio is 205 for this module as shown in figure 52, and reflects the dimensional constraints imposed by the fairing surface area. Optimum mixture ratio for Case 5 was 6.4 at an expansion ratio of 200. The high mixture ratio for Case 5 is typical of recoverable upper stage launch applications and reflects the importance of propellant bulk density on vehicle performance.

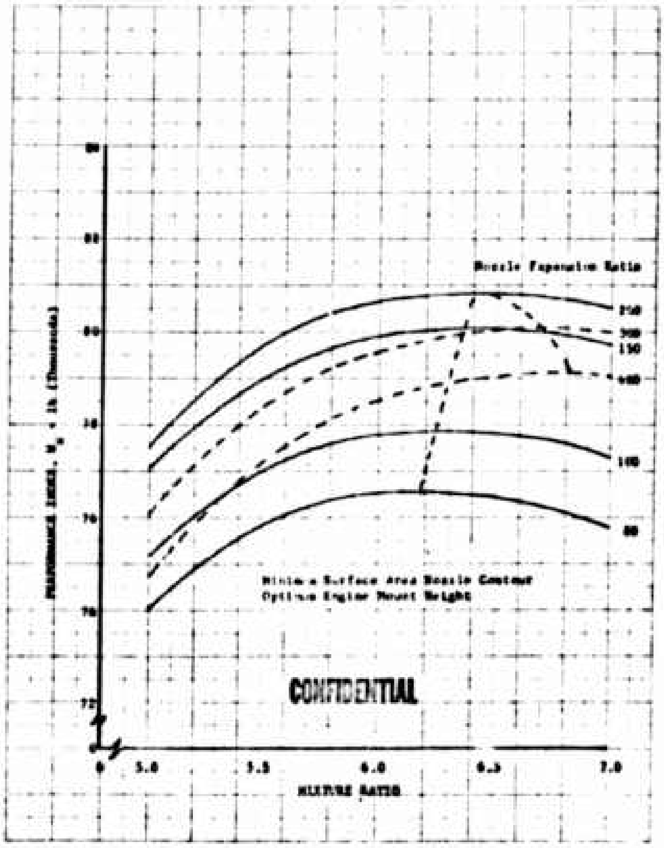


Figure 51. Performance Index vs Mixture Ratio for Case 5, 250K Module (ϵ_M = 250) DF 57564

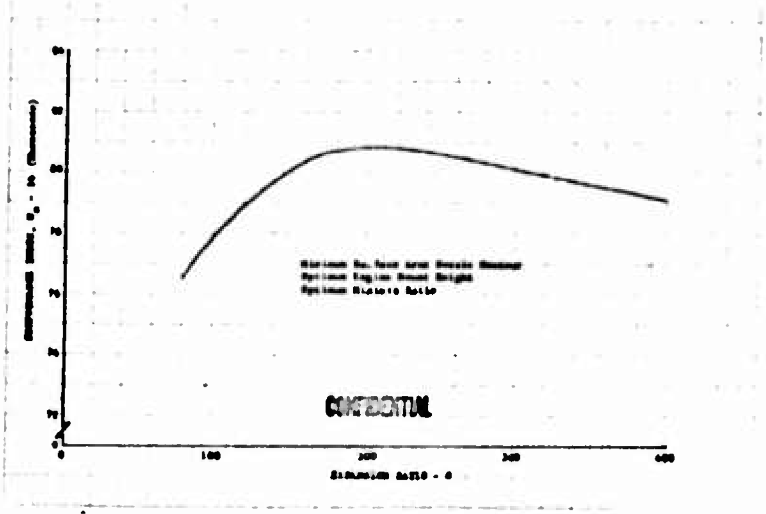


Figure 52. Performance Index vs Expansion Ratio for Case 5, 250K Module (ϵ_M = 250) DF 57565

f. Case 6 - Recoverable Upper Stage, Tandem

(C) The optimum (or maximum allowable due to the 250K limit) expansion ratio for Case 6 is equal to the module engine size. The performance as a function of mixture ratio and expansion ratio is shown in figure 53. Optimum mixture ratio for Case 6 is 6.7. Performance as a function of expansion ratio is shown in figure 54.

(U) The optimum engine expansion ratios for the various stages are quite different. Considerable vehicle performance gains can therefore accrue from matching the engine expansion ratio to the stage requirements. The quantitative performance advantages of stage matching is discussed in Section V.

D. COMMON MODULE SELECTION (350K MODULE)

1. Case Optimization

(U) The general Performance Index optimization and common module selection procedure outlined for the 250K module study was used in the 350K module study. The differences between 250K and 350K applications (Cases 1 through 6) are outlined in Appendix I. The basic differences between the applications for the two thrust classes are in vehicle dimensions, geometrical constraints, and number of engines specified for each case.

(U) The procedure used in selecting the 350K common module was the same as that used in selecting the 250K module.

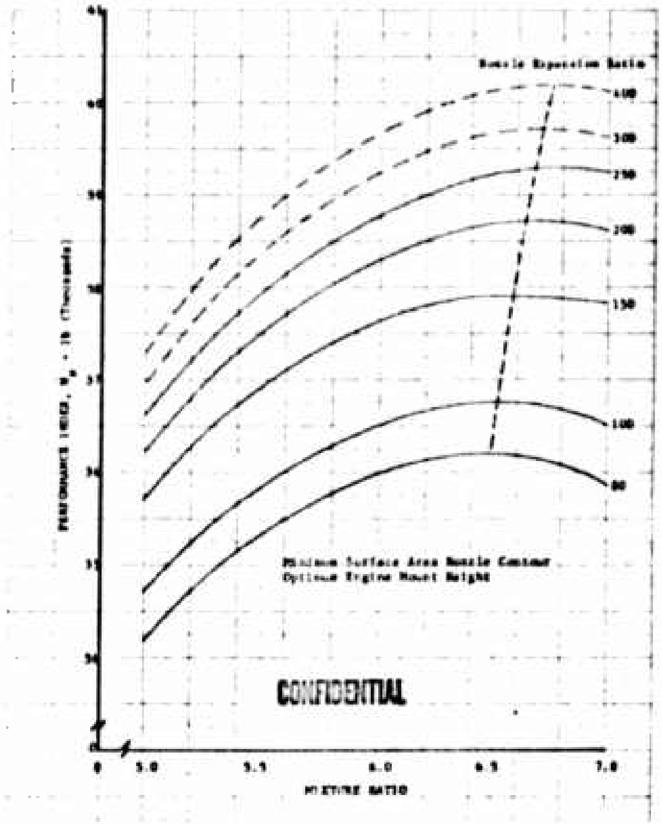


Figure 53. Performance Index vs Mixture Ratio for Case 6, 250K Module (ϵ_M = 250) DF 57574

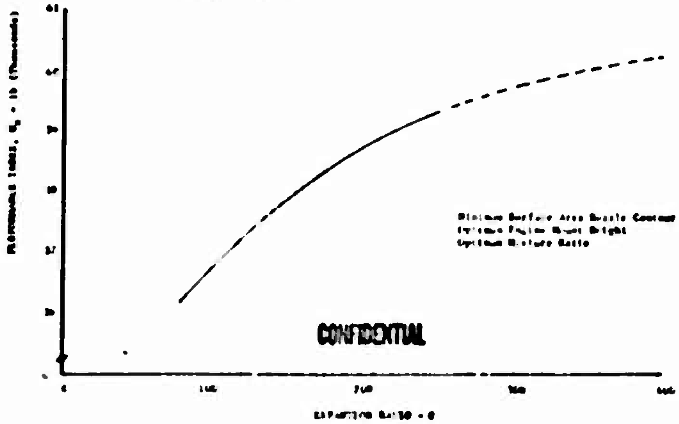


Figure 54. Performance Index vs Expansion Ratio for Case 6, 250K Module (ϵ_M = 250) DF 57575

(C) For the 350K module study, four basic engine module sizes ϵ_M = 100, 200, 300, and 400 were investigated. (The engine power module was sized to develop 350K at area ratios of 100, 200, 300, and 400 with minimum surface area nozzle contours.) The Performance Index of each application case was optimized for mixture ratio, expansion ratio, and engine mount position, using the same procedure as outlined in figures 12, 13, and 14.

(U) The optimization curves for each of the 350K cases are presented in the following paragraphs. Based on the results of the 250K study, the upper stage nozzle contours are minimum surface area and lower stage nozzle contours are maximum performance.

a. Case 1 - Expendable Lower Stage

(U) The expansion ratio optimization is shown in figure 55, and summary performance as a function of engine module size (ϵ_M) is shown in figure 56.

b. Case 2 - Expendable Upper Stage

(U) The 350K module Case 2 optimizes at the highest allowable expansion ratio and, therefore, the curves were generated as discussed in the 250K Case 2 module size selection. The expansion ratio-mixture ratio performance is shown in figure 57, and summary performance as a function of engine module size (ϵ_M) is shown in figure 58.

c. Case 3 - Single Stage-to-Orbit

(U) The expansion ratio optimization is shown in figure 59, and summary performance as a function of engine module size (ϵ_M) is shown in figure 60.

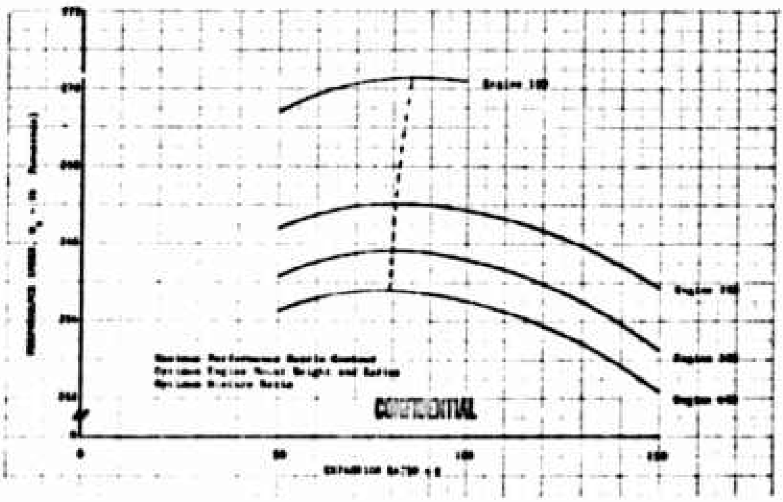


Figure 55. Performance Index vs Expansion Ratio for Case 1, 350K Module — DP 57615

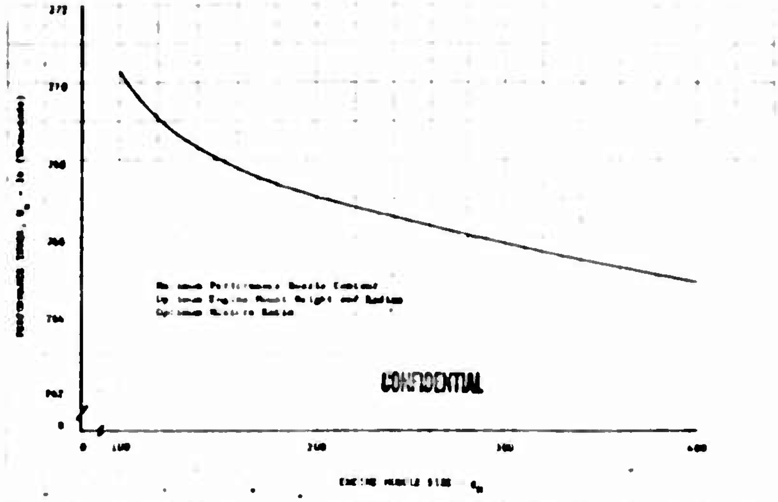


Figure 56. Performance Index vs Engine Module Size for Case 1, 350K Module   DF 57616

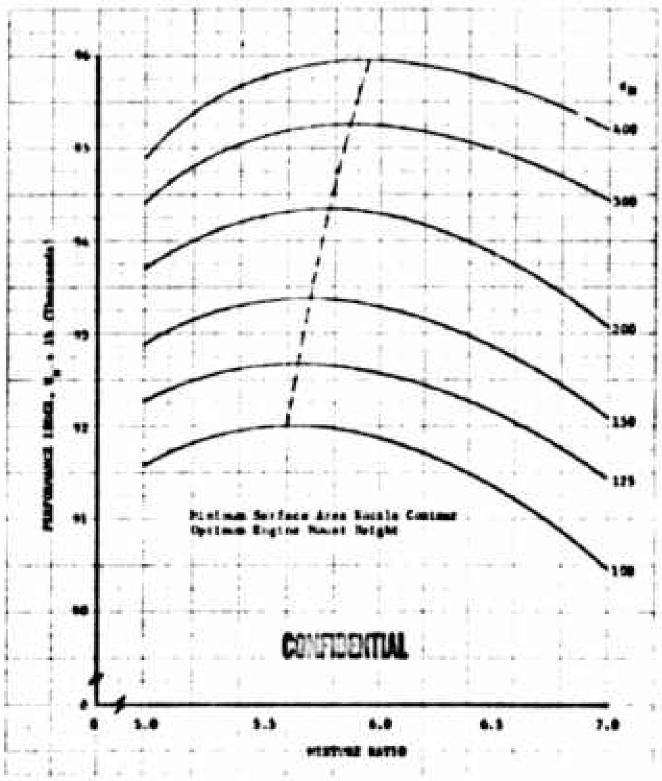


Figure 57. Performance Index vs Mixture Ratio for Case 2, 350K Module   DF 57617

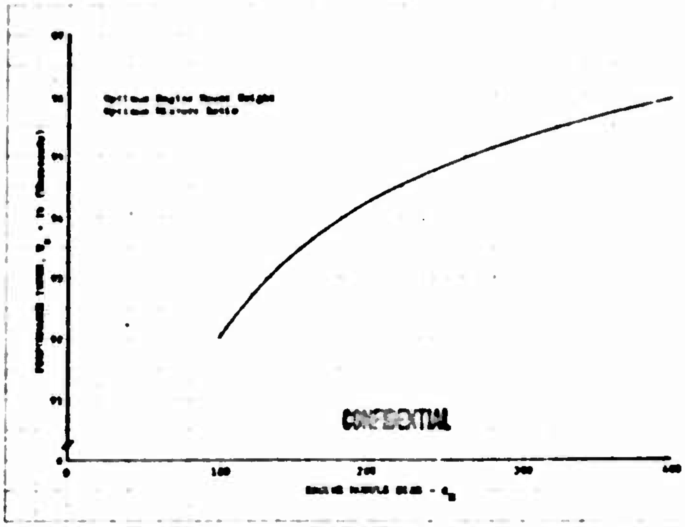


Figure 58. Performance Index vs Engine Module Size for Case 2, 350K Module DF 57618

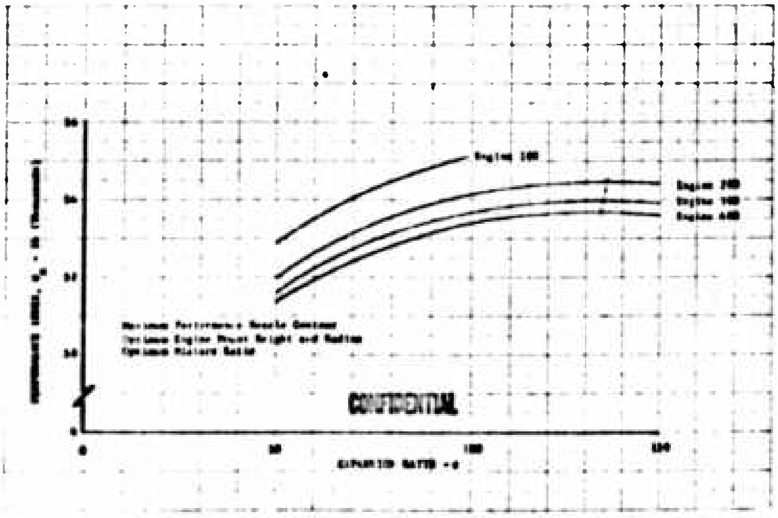


Figure 59. Performance Index vs Expansion Ratio for Case 3, 350K Module DF 57619

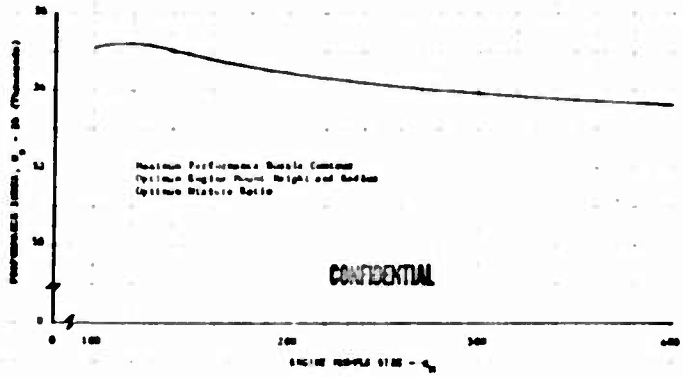


Figure 60. Performance Index vs Engine Module Size for Case 3, 350K Module   DF 57620

d. Case 4 - Recoverable Lower Stage

(U) The expansion ratio optimization is shown in figure 61, and summary performance as a function of engine module size (ϵ_M) is shown in figure 62.

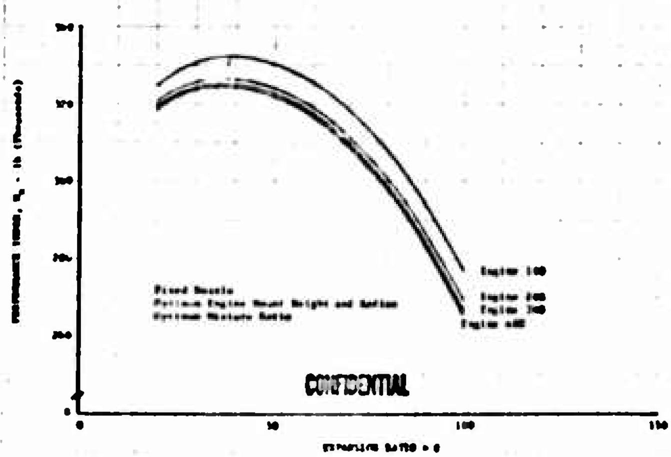


Figure 61. Performance Index vs Expansion Ratio for Case 4, 350K Module   DF 57621

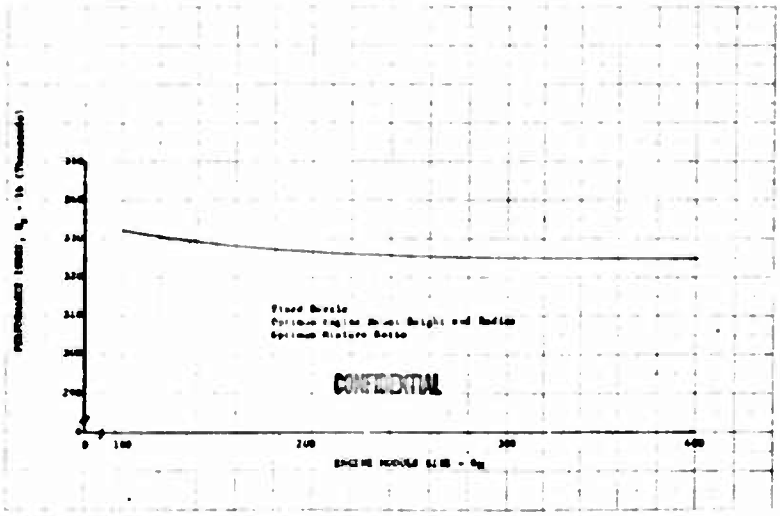


Figure 62. Performance Index vs Engine Module Size for Case 4, 350K Module DF 57622

e. Case 5 - Recoverable Upper Stage - Pick-a-Back

(U) The expansion ratio optimization is shown in figure 63, and summary performance as a function of module size (ϵ_M) is shown in figure 64.

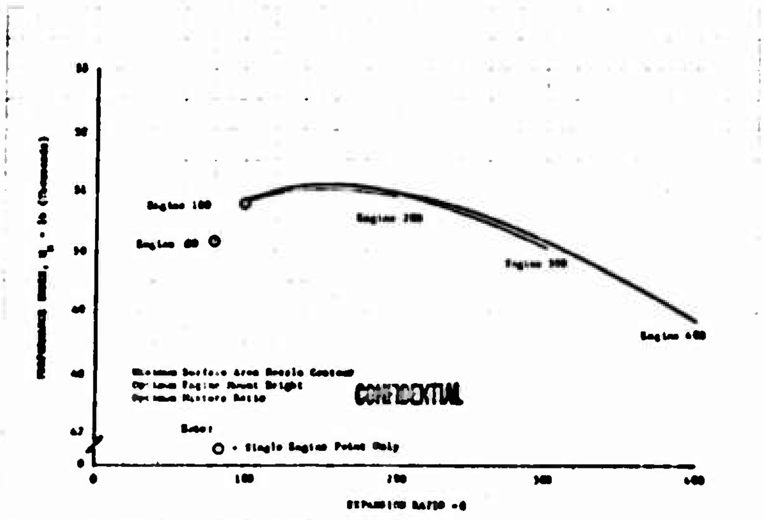


Figure 63. Performance Index vs Expansion Ratio for Case 5, 350K Module DF 57623

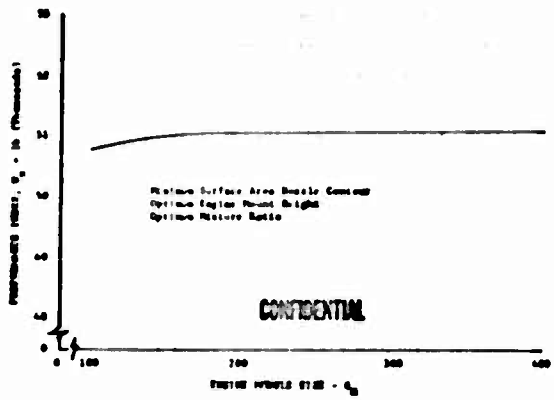


Figure 64. Performance Index vs Engine Module Size for Case 5, 350K Module DF 57624

f. Case 6 - Recoverable Upper Stage - Tandem

(U) The 350K module Case 6 optimizes at the highest allowable expansion ratio, and therefore, the curves were generated as discussed in the 250K Case 2 module size selection. The expansion ratio-mixture ratio performance is shown in figure 65, and summary performance as a function of module size (ϵ_M) is shown in figure 66.

2. Common Module Size

(C) Figure 67 gives the summary of the Performance Index as a function of engine module size. The percentage of the maximum Performance Index of each case is shown in the lower portion of figure 67. These curves are generated from figures 56, 58, 60, 62, 64, and 66 for Cases 1 through 6, respectively. These results indicate that the 350K cases are not as sensitive to flow rate as the 250K cases. (The average of the cases is shown in the upper portion of figure 67.) A common module delivering 350K vacuum thrust at an expansion ratio of 300 was selected.

3. Case Performance

(U) The performance of each case was established using power package size and flow rate determined from the above common engine module size selection analysis. The individual case optimization curves are given in the following paragraphs. The performance obtained is summarized in table II.

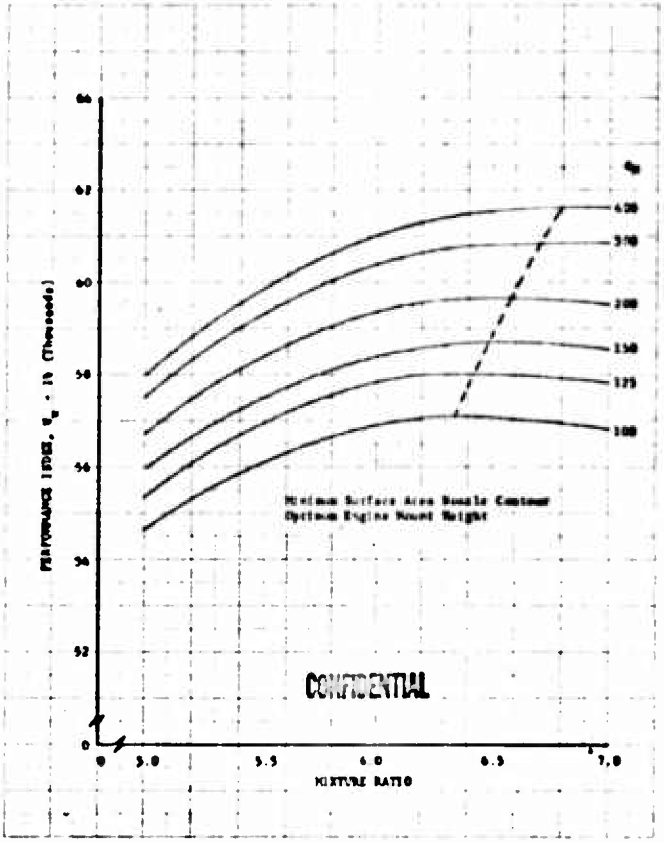
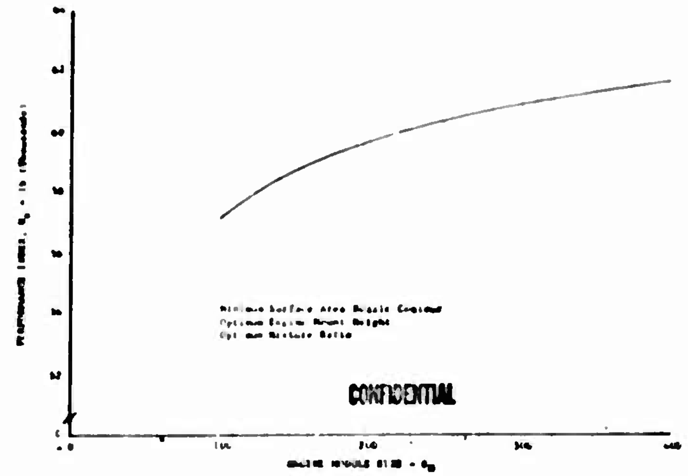


Figure 65. Performance Index vs Mixture Ratio for Case 6, 350K Module DF 57625



F'gure 66. Performance Index vs Engine Module Size for Case 6, 350K Module

DF 57626

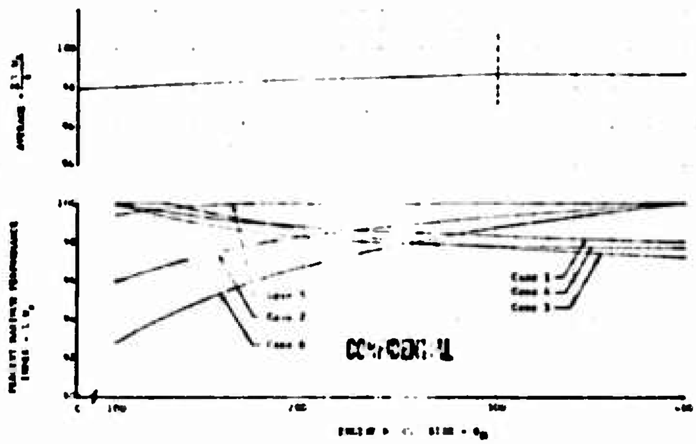


Figure 67. Percent Performance Index vs Engine Module Size, 350K Module

DF 57627

a. Case 1 - Expendable Lower Stage

(C) Performance as a function of mixture ratio and expansion ratio is shown in figure 68 for the common module size. Figure 69 is a cross-plot of these data to show optimum expansion ratio. Optimum expansion ratio in Case 1 is 80.

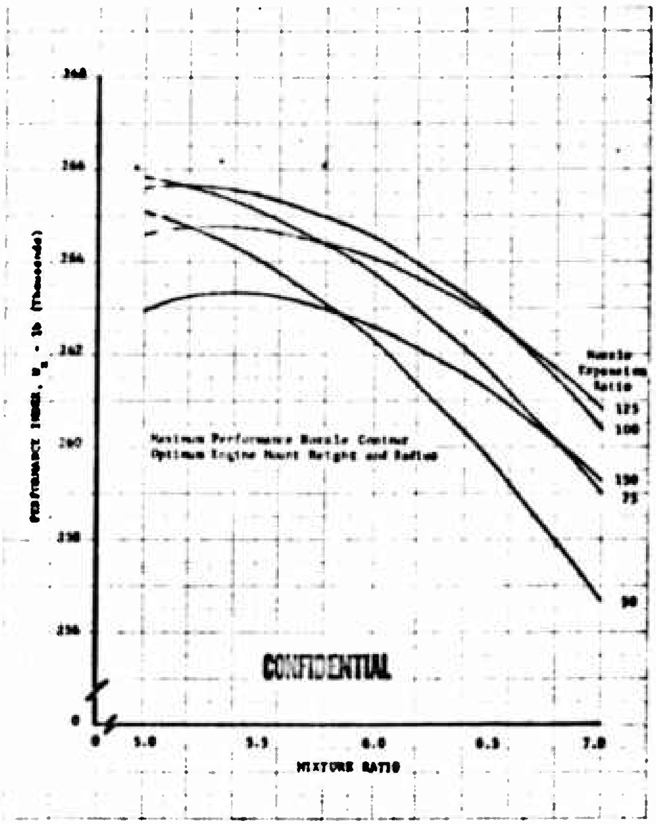


Figure 68. Performance Index vs Mixture Ratio for Case 1, 350K Module (ϵ_M = 300) DF 57628

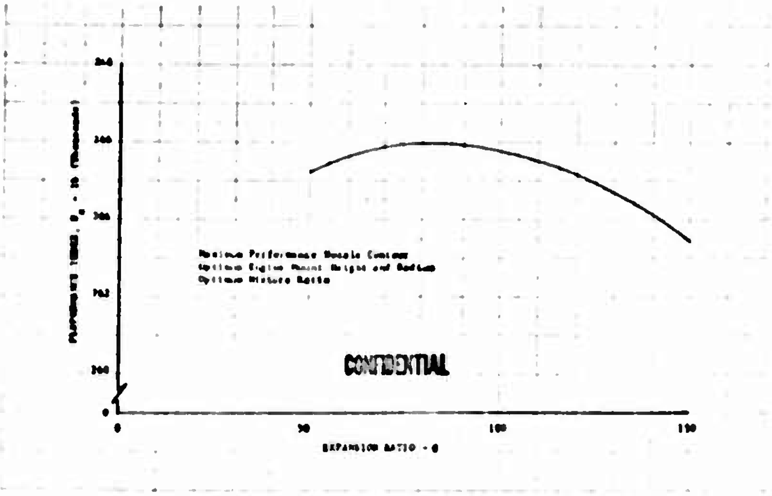


Figure 69. Performance Index vs Expansion Ratio for Case 1, 350K Module (ϵ_M = 300)

DF 57629

b. Case 2 - Expendable Upper Stage

(C) Performance as a function of mixture ratio and expansion ratio is shown in figure 70 for the common module size. Expansion ratio lines higher than an expansion ratio of 300 are shown dotted as these engines would deliver more than 350K vacuum thrust. Figure 71 is a cross-plot of these data to show optimum expansion ratio. This curve is also dotted above expansion ratios of 300, which is the highest expansion (and thus highest stage performance) that can be used consistent with the 350K thrust limit ground rule.

c. Case 3 - Single Stage-to-Orbit

(C) Performance as a function of mixture ratio and expansion ratio is shown in figure 72. Figure 73 is a cross-plot of these data to show optimum expansion ratio. Optimum expansion ratio for Case 3 is 135.

d. Case 4 - Recoverable Lower Stage

(C) Performance as a function of mixture ratio and expansion ratio is shown in figure 74 for the common module size. Figure 75 is a cross-plot of these data to show optimum expansion ratio. Optimum expansion ratio for Case 4 is 35.

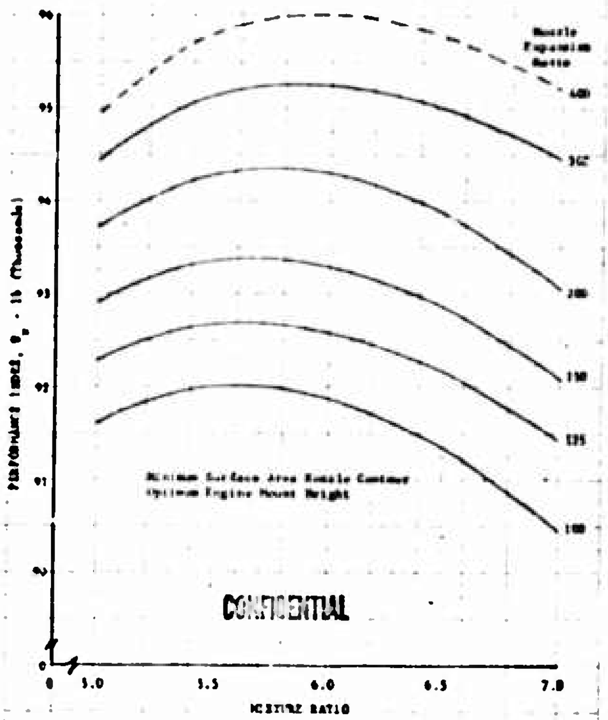


Figure 70. Performance Index vs Mixture Ratio for Case 2, 350K Module (ϵ_M = 300) DF 57630

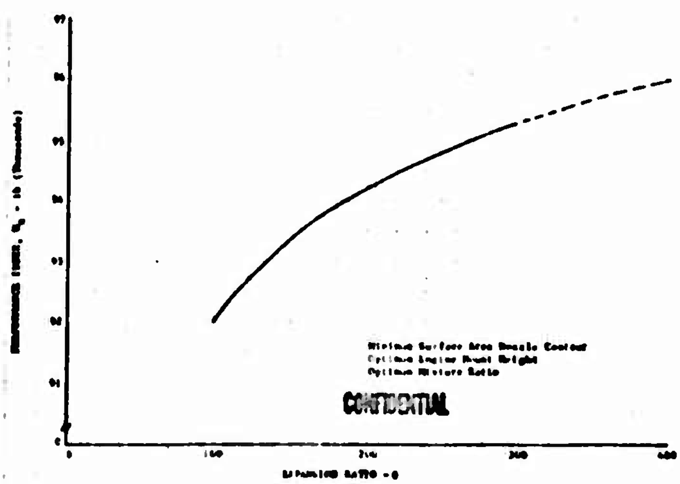


Figure 71. Performance Index vs Expansion Ratio for Case 2, 350K Module (ϵ_M = 300) DF 57631

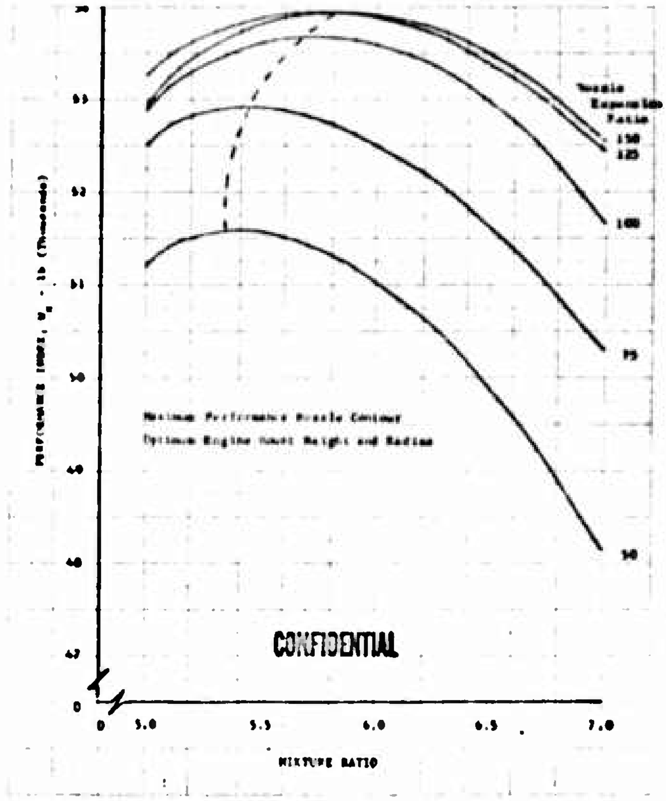


Figure 72. Performance Index vs Mixture Ratio for Case 3, 350K Module (ϵ_M = 300) DF 57632

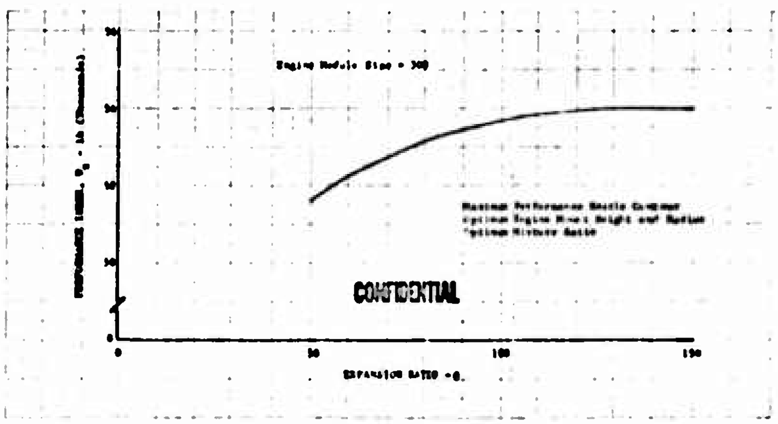


Figure 73. Performance Index vs Expansion Ratio for Case 3, 350K Module (ϵ_M = 300) DF 57633

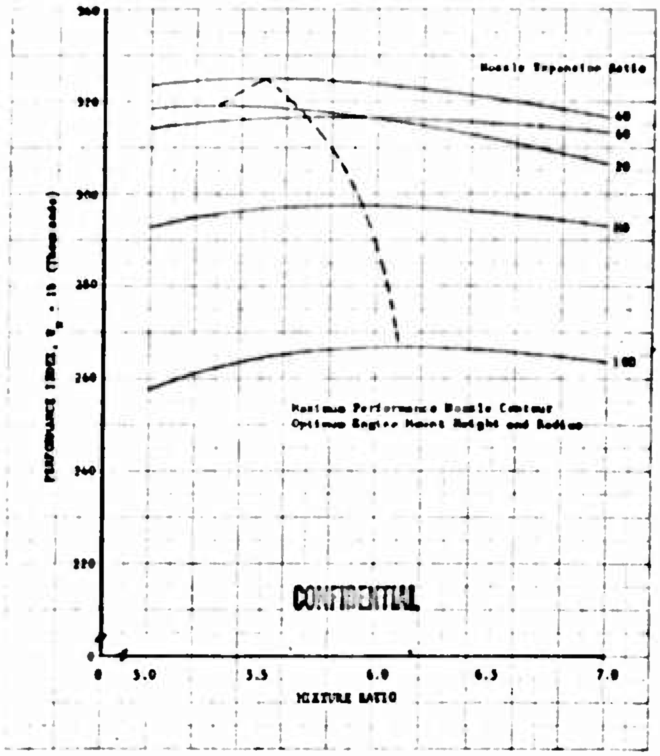


Figure 74. Performance Index vs Mixture Ratio for Case 4, 350K Module (ϵ_M = 300) DF 57634

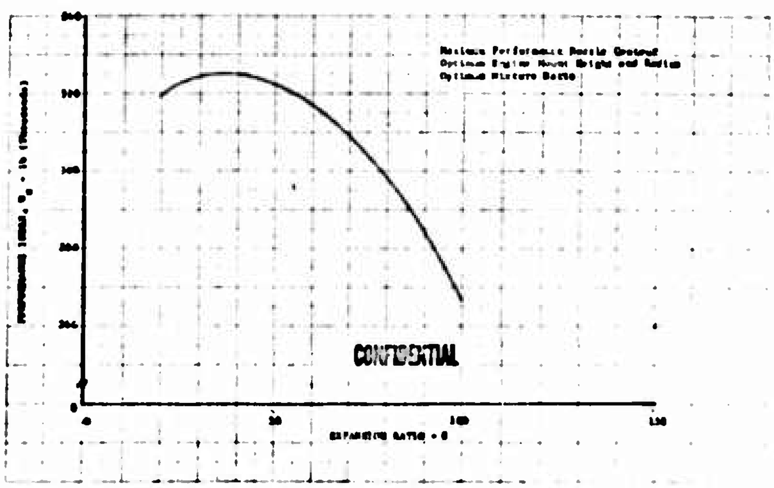


Figure 75. Performance Index vs Expansion Ratio for Case 4, 350K Module (ϵ_M = 300) DF 57635

e. Case 5 - Recoverable Upper Stage - Pick-a-Back

(C) Performance as a function of mixture ratio and expansion ratio is shown in figure 76 for the common module size. Figure 77 is a cross-plot of these data to show optimum expansion ratio. Optimum expansion ratio for Case 5 is 150. The relatively low expansion ratio of the 350K installation is because of the single engine mounted at the end of the tanks. This type of installation mounting results in increased surface fairing losses relative to 250K

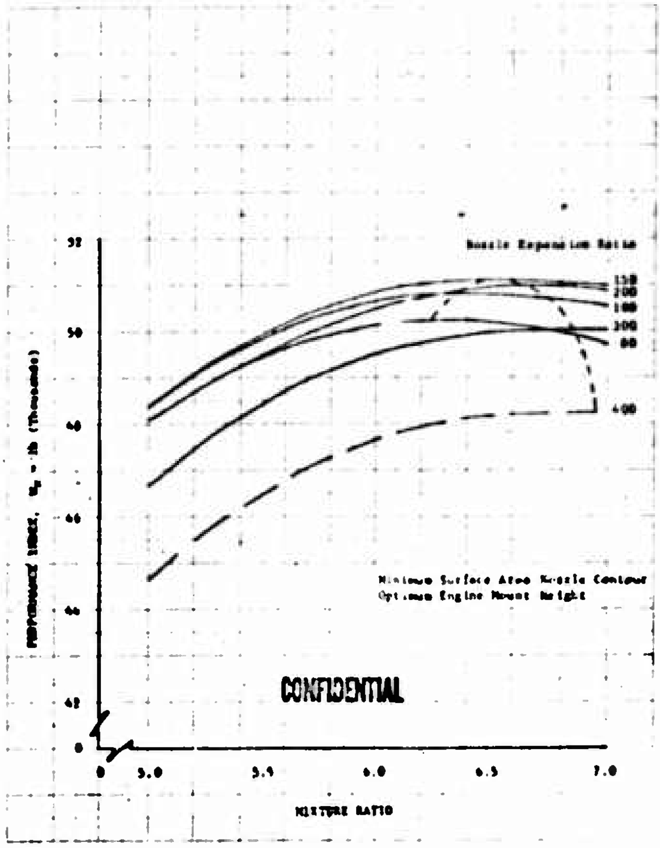


Figure 76. Performance Index vs Mixture Ratio for Case 5, 350K Module (ϵ_M = 300) DF 57636

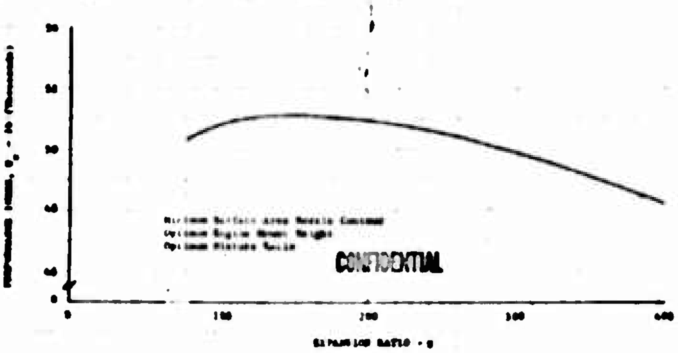


Figure 77. Performance Index vs Expansion Ratio for Case 5, 350K Module (ϵ_M = 300) DF 57637

f. Case 6 - Recoverable Upper Stage - Tandem

(C) Performance as a function of mixture ratio and expansion ratio is shown in figure 78 for the common module size. Figure 79 is a cross-plot of these data to show optimum expansion ratio. Lines on both of these figures are dotted for expansion ratios above 300 as was done with Case 2 because engine thrust would exceed 350K at higher expansion ratios. The highest Case 6 performance occurs at the highest permissible expansion ratio, which is 300.

E. EVALUATION OF NOZZLE CONTOURS (250K MODULE)

(U) Engine length and exhaust nozzle performance is a function of the exhaust nozzle contours. In general, shorter length contours yield lower nozzle performance. Therefore, in addition to optimization of the area ratio, the optimization of a bell nozzle engine includes the selection of the shape or contour of the nozzle. The bell nozzle contours used by Pratt & Whitney Aircraft are selected from a family of truncated perfect nozzles. Perfect nozzles are defined as those that, at a prescribed area ratio, expand a gas flow from the throat of the nozzle to a uniform and parallel flow at the nozzle exit. Using the method of characteristics, a series of perfect nozzle contours may be computed as a function of the design area ratio. The integrated thrust and surface area can also be calculated at axial locations along the nozzle. The calculation procedure includes the effect of friction and varying thermodynamic properties of the reacting gases. Representative results of this detailed analysis can be plotted as shown in figure 80 which presents coordinates for perfect bell nozzles with lines of constant surface area and constant vacuum thrust coefficients (which includes turning and frictional losses).

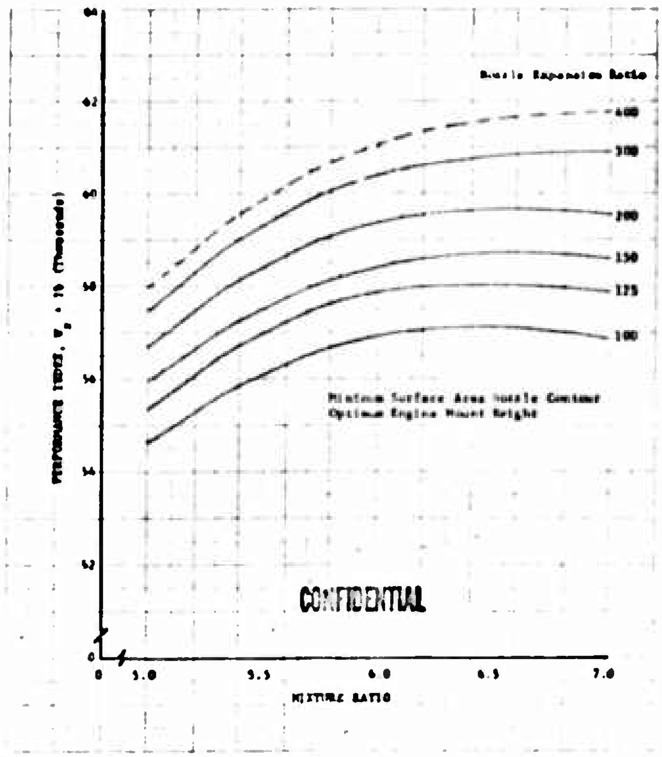


Figure 78. Performance Index vs Mixture Ratio for Case 6, 350K Module (ϵ_M = 300) DF 57638

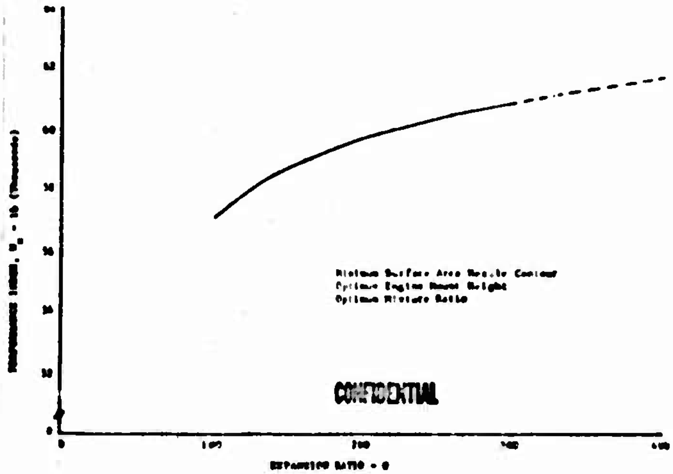


Figure 79. Performance Index vs Expansion Ratio for Case 6, 350K Module (ϵ_M = 300) DF 57639

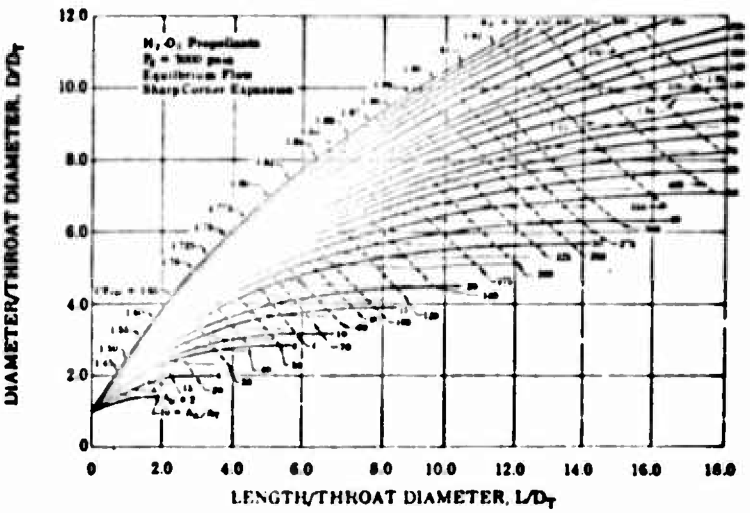


Figure 80. Perfect Nozzle Contours FD 23119

(U) A perfect nozzle contour does not necessarily produce optimum engine or vehicle performance. Because a perfect nozzle is constrained to produce completely axial flow at the exit, a considerable part of the aft section of the nozzle is involved in the final flow turning process. In a real nozzle the friction losses here are greater than the performance gains which accrue from the final flow straightening. Therefore, it is necessary to shorten or truncate a perfect nozzle to produce maximum nozzle performance. Further, in real vehicles there are engine length penalties such that additional truncation may be required to produce maximum vehicle performance. Pratt & Whitney Aircraft has thus defined a series of four decreasing nozzle length truncations of the above defined perfect nozzle for application to real vehicles, and each vehicle case in the application study was investigated to determine which nozzle truncation produced best vehicle performance. These truncations (referred to as nozzle contours in this report) are described as follows:

1. Maximum Performance Nozzle (MC$_s$)
2. Minimum Surface Area Nozzle (MSA)
3. Minimum Length (ML)
4. Base Nozzle.

These four nozzle truncations are shown schematically in figure 81.

(U) So that the method used in establishing MC$_s$, MSA, ML, and Base Nozzle contours can be more easily identified, a representative portion of the info mation given in figure 80 is shown in figure 82.

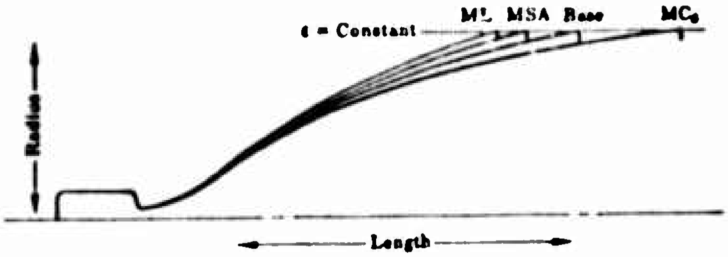


Figure 81. Nozzle Contour FD 22438

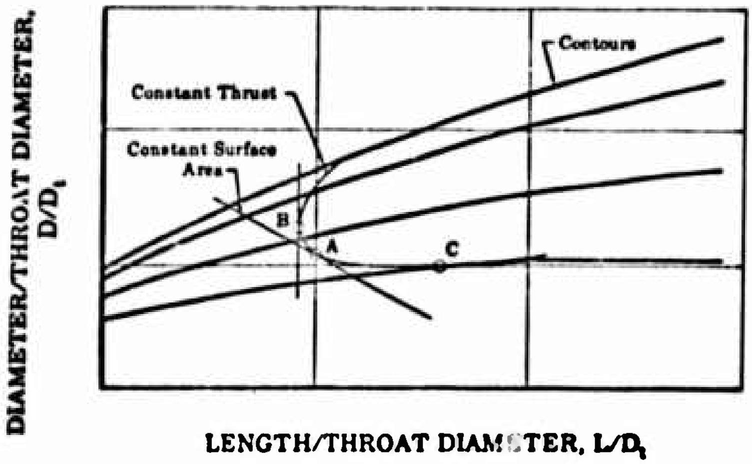


Figure 82. Contour Optimization FD 6263A

(U) Nozzles with minimum surface area (MSA nozzles) for a given thrust are defined by the locus of points for which a line of constant thrust coefficient ($C_{F_{vac}}$) and a line of constant surface area-to-throat area ratio (A_s/A_t) are tangent. This is shown as point A on figure 82. Nozzles with minimum length (ML contour) for a given thrust are defined by the locus of points for which lines of constant $C_{F_{vac}}$ are tangent to a vertical line (point B on figure 82). Maximum nozzle efficiency (MC_s) for a given thrust are defined by the locus of points for which the lines of constant thrust coefficient ($C_{F_{vac}}$) have zero slope (point C on figure 82). It should be noted that the maximum performance nozzle (point C) is still short of the full length perfect nozzle because frictional drag has been included.

(U) A fourth type of nozzle truncation considered is referred to as a "Base" nozzle contour. This truncation has resulted from experience with various optimization studies and generally produces nearly optimum performance, particularly for lower stage applications. While this contour is not established directly from analysis of figure 82, it falls approximately half-way between points A (MSA) and C (MC_s).

(U) Detailed studies of the effect of nozzle contour were conducted early in the program. These results, which were based on analysis using the fixed regeneratively cooled nozzle, indicated the use of minimum surface area nozzles for the upper stages and maximum performance nozzle contours for the lower stages. The nozzle contours studied were minimum length (ML), minimum surface area (MSA), base, and maximum performance (MC_s). The nozzle contour analysis was re-evaluated for the two-position lightweight nozzle and the same results for upper and lower stages were indicated.

(U) The percent of maximum Performance Index values for each case using the fixed nozzle are shown in table VII. A common engine module was obtained for each contour, which is also noted in the table. The total upper stage performance is approximately equal for the ML, MSA and base nozzles. Because the MSA contour was optimu for two of the three upper stage cases, this contour was selected for the upper stage engine. The result of the lower stage nozzle contour investigation is shown in table VIII as a percent of maximum performance for each lower stage case. The values are for the common module flow rate obtained by using the minimum surface area upper stages.

(U) The results for the two-position nozzle are shown in table IX for the upper stages. There is little difference, but again the minimum surface area nozzle contour was chosen. For the lower stages, maximum performance and base nozzles were considered. The optimum case expansion ratio varies with the contour for a given module flow rate. The evaluation is shown in figures 83, 84, and 85 for Cases 1, 3, and 4, respectively. These results are based on using the common module flow rate. The maximum performance nozzle indicates higher performance and slightly lower optimum case area ratio. Performance is lower with this nozzle contour at higher area ratios due to the increased length (relative to the base contour), which results in higher skirt drag. The maximum performance nozzle contour shows a slight performance advantage as compared to the base nozzle contour and was therefore selected. In a real vehicle optimization, it may be possible that other considerations (such as a restriction on overall vehicle length) might influence the nozzle contour selection in favor of a shorter length nozzle such as the base contour.

(C) Table VII. Performance Index Nozzle Contour Summary for Upper Stages (Percent of Maximum) Fixed Regenerative Nozzle

Case	ML (ϵ_M = 200)	MSA (ϵ_M = 181)	Base (ϵ_M = 164)	MC_s (ϵ_M = 130)
2	99.89	99.92	100.00	99.28
5	99.93	100.00	99.76	98.91
6	99.86	100.00	99.97	99.05

Note: ϵ_M = Upper stage common engine module area ratio.

(C) Table VIII. Performance Index Nozzle Contour Summary for Common Engine Module, Lower Stages (Percent of Maximum) Fixed Regenerative Nozzle

Case	MC_s	Base	MSA	ML
1	100.00	99.58	98.81	97.28
3	100.00	98.75	96.62	92.83
4	100.00	99.37	98.64	96.86

Note: Engine flow rate based on ϵ_M = 181.

(U) Table IX. Performance Index Nozzle Evaluation, Two-Position Lightweight Nozzle (Percent of Maximum)

Case	MSA	Base	MC_s
Case 2	99.85	100.00	99.85
Case 5	100.00	97.78	98.14
Case 6	100.00	99.88	99.81

(U) A fourth type of nozzle truncation considered is referred to as a "Base" nozzle contour. This truncation has resulted from experience with various optimization studies and generally produces nearly optimum performance, particularly for lower stage applications. While this contour is not established directly from analysis of figure 82, it falls approximately half-way between points A (MSA) and C (MC_s).

(U) Detailed studies of the effect of nozzle contour were conducted early in the program. These results, which were based on analysis using the fixed regeneratively cooled nozzle, indicated the use of minimum surface area nozzles for the upper stages and maximum performance nozzle contours for the lower stages. The nozzle contours studied were minimum length (ML), minimum surface area (MSA), base, and maximum performance (MC_s). The nozzle contour analysis was re-evaluated for the two-position lightweight nozzle and the same results for upper and lower stages were indicated.

(U) The percent of maximum Performance Index values for each case using the fixed nozzle are shown in table VII. A common engine module was obtained for each contour, which is also noted in the table. The total upper stage performance is approximately equal for the ML, MSA and base nozzles. Because the MSA contour was optimum for two of the three upper stage cases, this contour was selected for the upper stage engine. The result of the lower stage nozzle contour investigation is shown in table VIII as a percent of maximum performance for each lower stage case. The values are for the common module flow rate obtained by using the minimum surface area upper stages.

(U) The results for the two-position nozzle are shown in table IX for the upper stages. There is little difference, but again the minimum surface area nozzle contour was chosen. For the lower stages, maximum performance and base nozzles were considered. The optimum case expansion ratio varies with the contour for a given module flow rate. The evaluation is shown in figures 83, 84, and 85 for Cases 1, 3, and 4, respectively. These results are based on using the common module flow rate. The maximum performance nozzle indicates higher performance and slightly lower optimum case area ratio. Performance is lower with this nozzle contour at higher area ratios due to the increased length (relative to the base contour), which results in higher skirt drag. The maximum performance nozzle contour shows a slight performance advantage as compared to the base nozzle contour and was therefore selected. In a real vehicle optimization, it may be possible that other considerations (such as a restriction on overall vehicle length) might influence the nozzle contour selection in favor of a shorter length nozzle such as the base contour.

(C) Table VII. Performance Index Nozzle Contour Summary for Upper Stages (Percent of Maximum) Fixed Regenerative Nozzle

Case	ML (ϵ_M = 200)	MSA (ϵ_M = 181)	Base (ϵ_M = 164)	MC_s (ϵ_M = 130)
2	99.89	99.92	100.00	99.28
5	99.93	100.00	99.76	98.91
6	99.86	100.00	99.97	99.05

Note: ϵ_M = Upper stage common engine module area ratio.

(C) Table VIII. Performance Index Nozzle Contour Summary for Common Engine Module, Lower Stages (Percent of Maximum) Fixed Regenerative Nozzle

Case	MC_s	Base	MSA	ML
1	100.00	99.58	98.81	97.28
3	100.00	98.75	96.62	92.83
4	100.00	99.37	98.64	96.86

Note: Engine flow rate based on ϵ_M = 181.

(U) Table IX. Performance Index Nozzle Evaluation, Two-Position Lightweight Nozzle (Percent of Maximum)

Case	MSA	Base	MC_s
Case 2	99.85	100.00	99.85
Case 5	100.00	97.78	98.14
Case 6	100.00	99.88	99.81

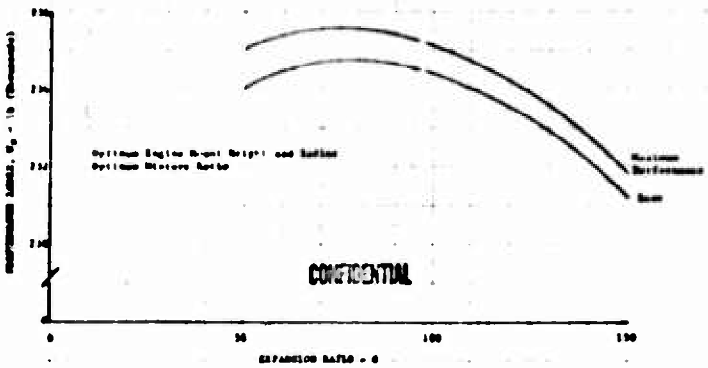


Figure 83. Nozzle Evaluation for Case 1, 250K Module (ϵ_M = 250) DF 57554

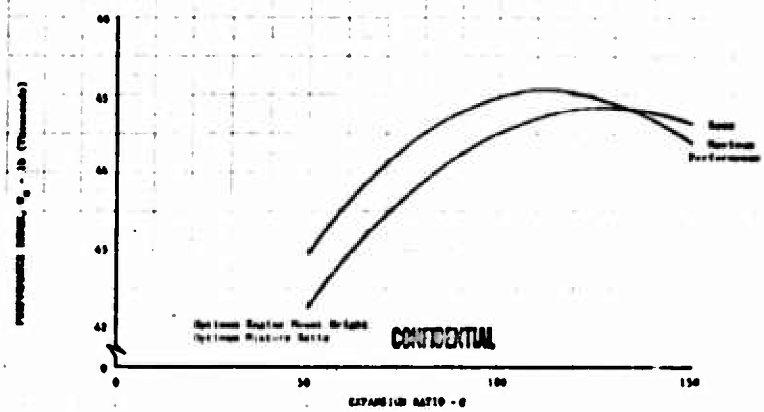


Figure 84. Nozzle Evaluation for Case 3, 250K Module (ϵ_M = 250) DF 57555

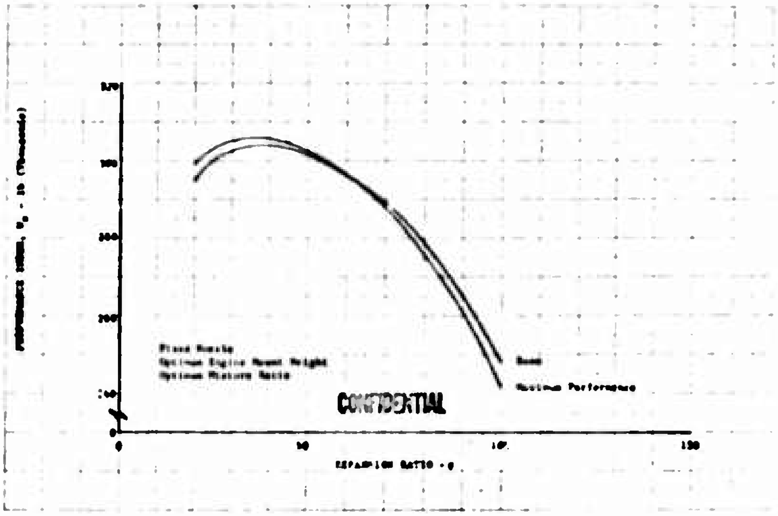


Figure 85. Nozzle Evaluation for Case 4, 250K Module (ϵ_M = 250) DF 57556

F. EVALUATION OF NOZZLE TRANSLATION POINT AREA RATIO (250K MODULE)

(C) Performance Index (W_x) values reported for the lower stage cases are based on engines with two-position nozzles that translate from an area ratio of 35 (primary area ratio). Increasing the area ratio at which translation occurs (i.e., increasing the area ratio of the fixed portion of the nozzle) results in slightly reduced sea level performance but slightly higher impulse at the translation altitude. This trend is illustrated in figure 86, which gives impulse for primary area ratios of 35, 50, and 80 for a typical overall area ratio of 100. Because upper stage cases do not operate in the atmosphere (or with the nozzle retracted), the point at which the translation occurs does not affect performance and thus is selected only for minimum stowed engine length.

(C) A tradeoff study was conducted to evaluate the optimum primary area ratio for lower stage Cases 1 and 3. As discussed previously, lower stage Case 4 is diameter limited and tends to optimize at low overall engine area ratios Case 1 and 3 were optimized using primary area ratios of 35, 50, and 80 as shown in figure 87. Optimum performance index values were obtained with primary area ratios in the 35 to 40 range. Engine module flow rate was held constant for this optimization.

(C) In conclusion, a translation point area ratio of 35 was selected for Cases 1 and 3.

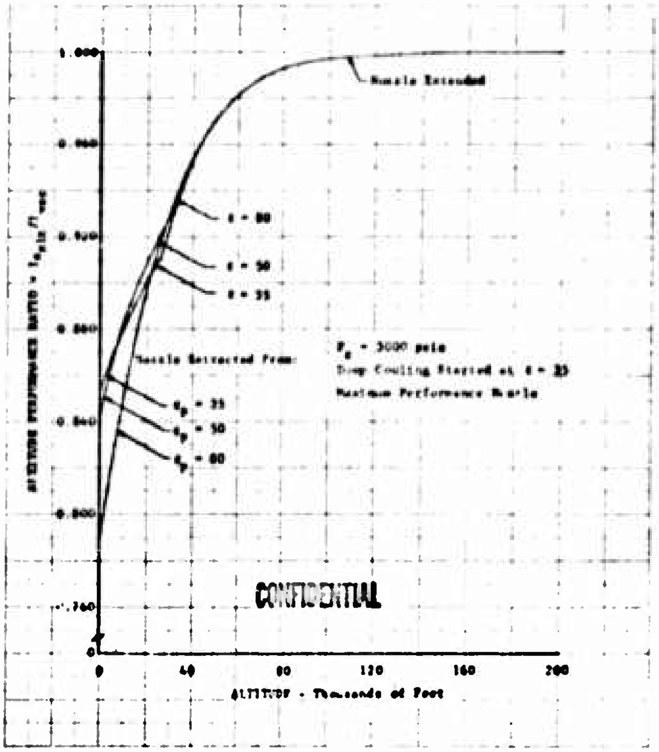


Figure 86. Single Engine Altitude Performance Ratio, Two-Position Nozzle-Area Ratio of Fully Extended Nozzle = 100 (r = 6.0) DF 57646

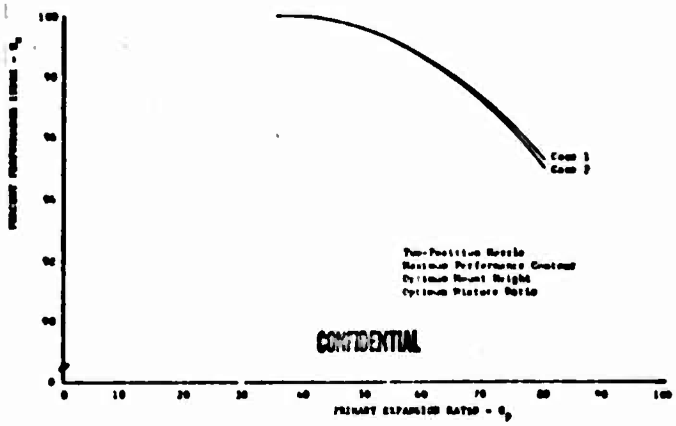


Figure 87. Percent Performance Index vs Primary Expansion Ratio DF 57568

G. EVALUATION OF DESIGN CHAMBER PRESSURE (250K MODULE)

(C) An evaluation of the effects of the engine design chamber pressure level on the performance of the Application Study cases was made. Chamber pressure was evaluated by determining a common engine module for various chamber pressure levels. Each application case was optimized similar to the previously discussed procedures. Chamber pressure of 2000, 2500, 3000, and 3500 psia were evaluated for each case for the 250K module. The two-position lightweight nozzle engine configuration was used in this analysis. The engine performance used in this analysis is based on the assumption of regenerative cooling at 2000 psia. As the chamber pressure was increased from 2000 psia to 3500 psia, transpiration cooling was used with the amount of transpiration cooling flow increased in direct proportion to the chamber pressure.

(U) Because chamber pressure reflects basic design changes in the power package and module throat design area, the evaluation of chamber pressure for Cases 1 through 6 was conducted for one nozzle contour.

(C) The results of this evaluation are shown in figure 88 for all of the application cases. These data are presented as a percent of the maximum performance (for each case) so that all cases may be compared. The upper portion of figure 88 represents a direct average of the six cases. It is shown that the vehicle average optimum chamber pressure is in excess of 3000 psia.

(C) The lower stages, and the size constrained Case 5 upper stage, show performance improvement with increased chamber pressure. For the lower stage, high chamber pressure provides high performance during the critical low altitude portion of the trajectory. This is due to the reduced exhaust area of the nozzle, which in turn reduces the associated thrust loss when ambient pressure acts on the exhaust area of the nozzle. Case 3 indicates the steepest performance gradient with chamber pressure and shows the high sensitivity of the single stage-to-orbit missions to increased engine performance. Higher chamber pressure improves sea level performance and also allows packaging of higher area ratios within vehicle diameter limits, which improves vacuum performance.

(U) Case 4 has a relatively steep slope due to engine length and diameter constraints because the size of the staged combustion module decreases with increasing chamber pressure. Higher chamber pressures permit a more compact cluster installation arrangement for Case 4.

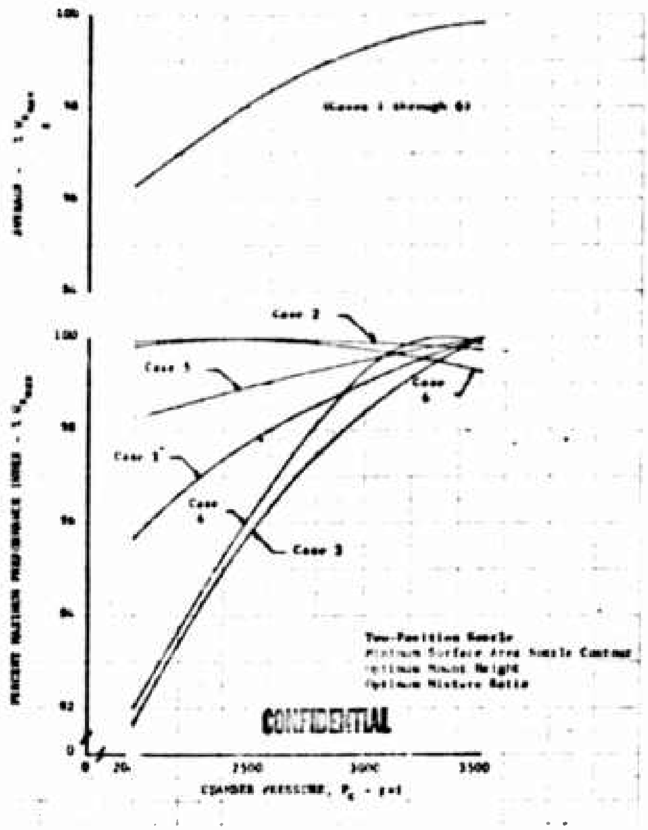


Figure 88. Percent Performance Index vs Chamber Pressure DF 57567

(C) The recoverable upper stage application (Case 5) indicates improved performance with increasing chamber pressure. The high chamber pressure for the upper stage application reflects the engine length constraint of this application. Cases 2 and 6 are relatively insensitive to chamber pressure over the range considered. As noted in the previous optimization analysis, Cases 2 and 6 are not size sensitive.

(C) As shown on the upper curve of figure 88, the vehicle average optimum chamber pressure is higher than 3000 psia. The present upper limit of engine technology is approximately 3000 psia, therefore, 3000 psia was selected as the chamber pressure for engines used in the application studies.

H. INSTALLATION LAYOUTS (250K MODULE)

(U) The optimum engine locations for the performance index optimization were determined by the parametric installation studies using analytical methods. To more fully assess the potential component interference and

provide a final installation weight, layouts were made of the engine installations for each 250K application case. The layouts resulted in only small weight changes from the parametric analysis. The final installation weights for the six application cases are shown in table X along with corresponding values from the parametric data in Section VI. The final installation weights shown in table X were used in determining the performance index values in table I. The installation dimensional nomenclature is illustrated in figure 89.

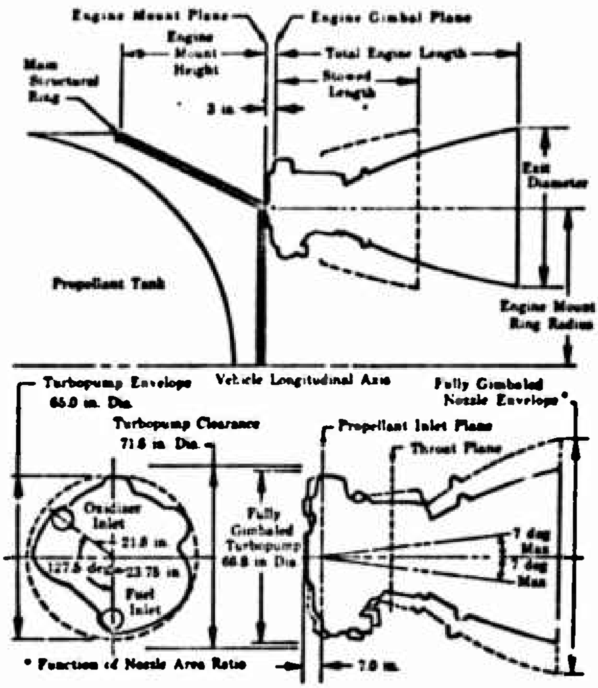


Figure 89. Typical Multiengine Installation FD 23129

(U) Table X. Installation Weights

Case	1	2	3	4	5	6
Mount Height, in.	88	73	80	47	52	45
Mount Radius, in.	101	—	118	98	—	—
Component Weights, lb						
Thrust Structure	2190 (-30)*	1285 (0)	2610 (-30)	2335 (-60)	2500 (0)	1000 (0)
Oxygen Fuel Lines	383 (-1)	55 (0)	383 (0)	417 (-64)	143 (0)	55 (0)
Hydrogen Fuel Lines	299 (-17)	55 (0)	300 (0)	551 (11)	136 (0)	55 (0)
Pressurization System	152 (-2)	24 (0)	145 (-6)	233 (-23)	48 (0)	24 (0)

*Numbers in parantheses reflect weight change from parametric data

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Hydrogen Fuel Lines	299 (-17)	55 (0)	300 (0)	551 (11)	136 (0)	55 (0)
Pressurization System	152 (-2)	24 (0)	145 (-6)	233 (-23)	48 (0)	24 (0)

*Numbers in parantheses reflect weight change from parametric data

(C) The Case 1 installation layout, figure 90, shows five engines mounted in a cruciform arrangement with the necessary installation hardware. The mounting position of the heat shield was set to provide sufficient ventilation airflow for the secondary nozzle. The performance index optimization studies resulted in a module with an area ratio of 75, a mount ring radius of 101 inches, and a mount height of 78 inches. It was necessary to increase the mount height on the layout to 88 inches to provide the required clearance between the center engine propellant feed lines and the propellant tank. The mount radius was held constant to provide clearance between adjacent engines in their fully gimbaled (± 7 degrees) positions. The cone within a cone thrust structure arrangement used in the parametric studies was retained. A slight change in the center engine crossbeams was made to provide a more uniform distribution of radial loads and a more direct feed line routing. The parametric design analysis considered crossbeams running from opposite engines with the center engine mounted at the beam intersections. The layout analysis, figure 90, indicated that a smaller center engine crossbeam was required, in addition to a more uniformly distributed radial loading if the cone stabilizing beams were tied into the center engine crossbeams rather than extending the full gimbal mount diameter as used in the parametric study. This resulted in a thrust structure weight decrease of 30 pounds. The propellant feed line weights decreased 18 pounds because of the plumbing rerouting. The pressurization system lines considered in the parametric study extended from the engines to a manifold near the center of the vehicle. This routing used on the layout resulted in a weight reduction of 2 pounds. The remaining installation components are the same as those used in the parametric study.

(C) The Case 2 installation layout, figure 91, shows the single engine and installation hardware. The performance index optimization studies resulted with an engine area ratio of 250 and a mount height of 73 inches. The installation concepts used in the parametric analysis were retained and there was no interference or design changes required. The resulting installation weights were the same as for the parametric studies.

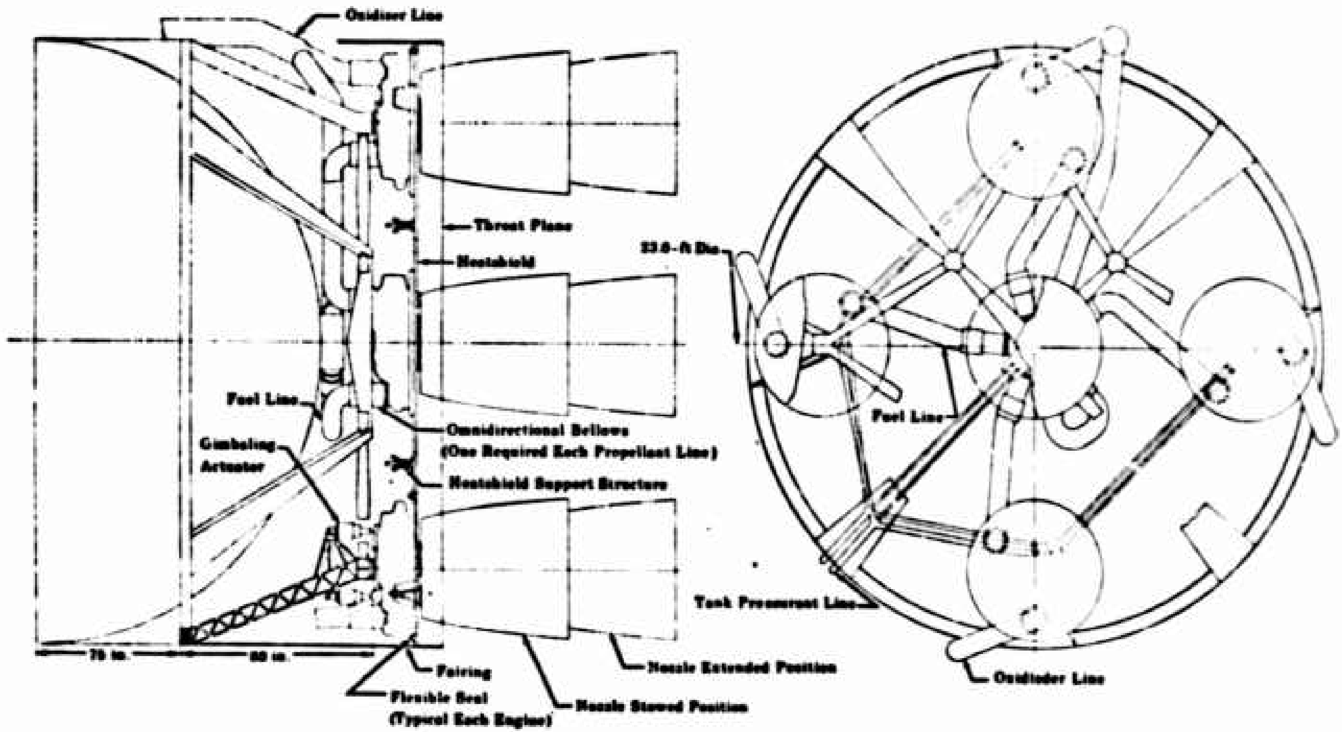

Figure 90. Case 1 Installation Layout

FD 22838

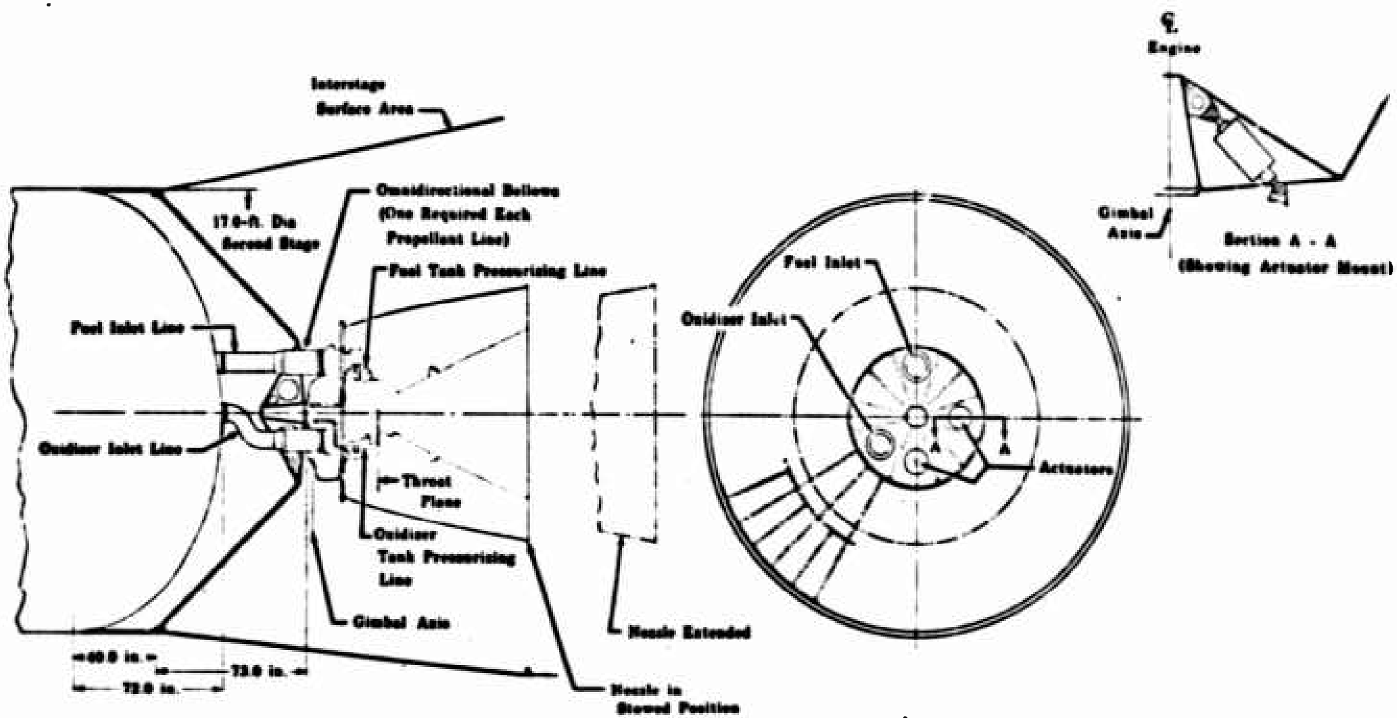

Figure 91. Case 2 Installation Layout

FD 22818

(C) The Case 3 installation layout, shown in figure 92, is similar to the one described above for Case 1. This case is also a five engine cruciform installation using the same design concepts selected for the Case 1 layout. The mounting position of the heat shield was set to provide sufficient ventilation airflow for the secondary nozzle. The Performance Index optimization resulted with an engine area ratio of 110, a mount ring radius of 118 inches, and a mount height of 80 inches. In this case, there was no interference on mount height because of the increased clearance of the elliptical hydrogen tank dome. The only weight change of the layout installation was due to the design change described in the Case 1 discussion on the more uniform distribution of radial loads and the pressurization system line weights. These changes resulted in a thrust structure weight decrease of 30 pounds and a pressurization system weight decrease of 6 pounds. The remaining installation components are the same as those used in the parametric studies.

(C) The installation layout for Case 4, figure 93, shows an eight engine cluster arranged in a circular pattern. The optimum conditions selected by the performance index optimizations were an engine area ratio of 35, a gimbal mount radius of 98 inches and a mount height of 47 inches. The feed line weights decreased 53 pounds due to plumbing rerouting. The pressurization system for the Case 4 parametric line routings shown on the layout resulted in a pressurization system weight decrease of 23 pounds from the parametric data. The thrust structure weights used in the parametric study were based on engine thrust levels of 250K and 350K. Because the vacuum thrust of each respective engine with area ratios of 35 is 237.0K and 331.8K, the thrust structures were resized for these values. The thrust structure weight decrease was 60 pounds for the 250K engine and 85 pounds for the 350K engine. The remaining installation components are the same as those used in the parametric studies.

(C) The Case 5 installation layout is shown in figure 94. This installation consists of two engines mounted at 78 inches on each side of the vehicle centerline. The optimum conditions selected by the Performance Index studies were an engine area ratio of 200 and a mount height of 52 inches. There were no interference problems and the installation weight is the same as used in the parametric studies.

(C) The installation layout for Case 6 is shown in figure 95. This installation is a single engine mounted on the vehicle centerline. The optimum installation conditions determined by the Performance Index studies were an engine area ratio of 250 and a mount height of 33 inches. The layout mount height was increased to 45 inches due to interference between the oxygen feed line and the thrust structure crossbeam. The increased mount height eliminated the interference due to the reduction in the crossbeam size with increased mount height. The installation components are the same as those used in the parametric studies.

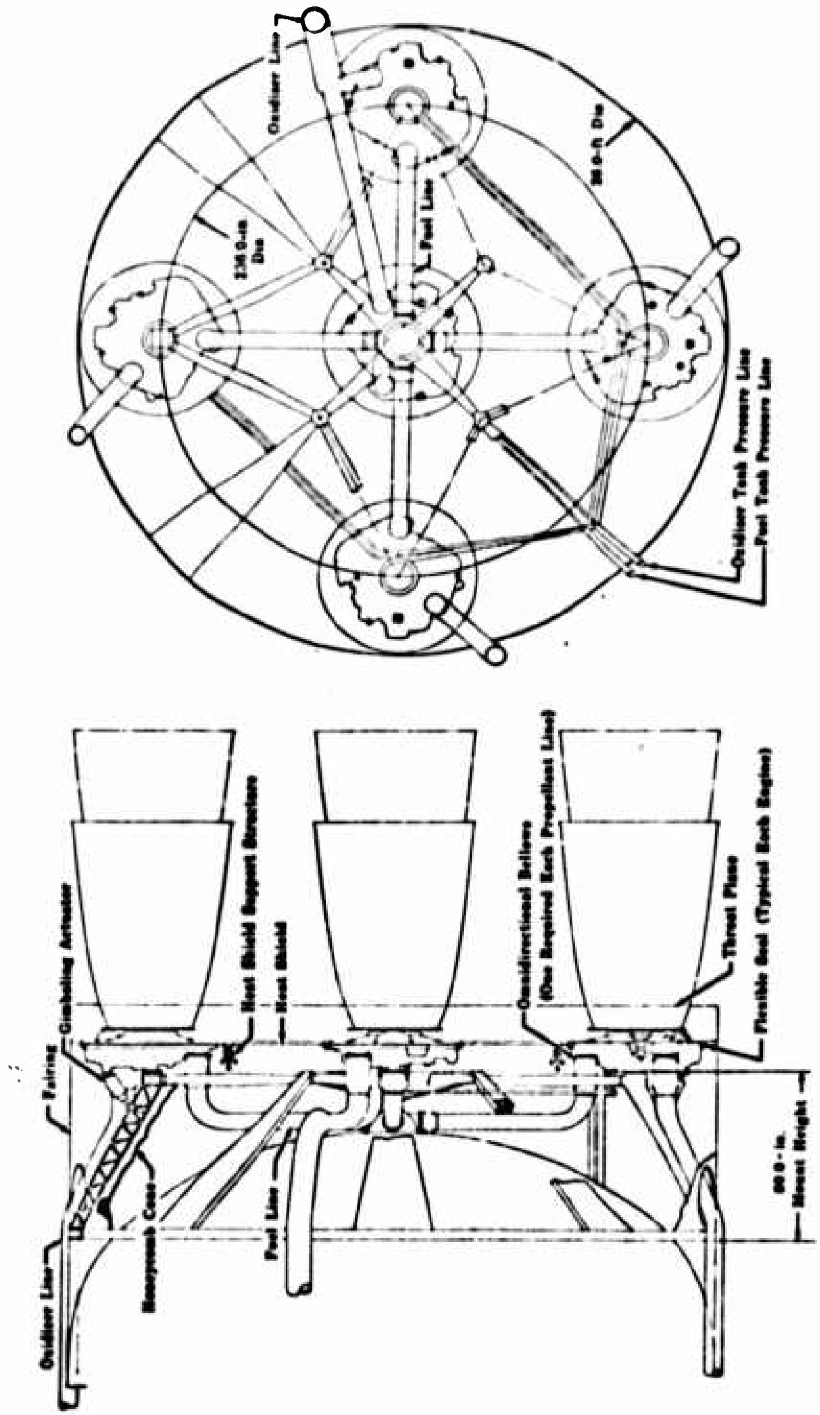

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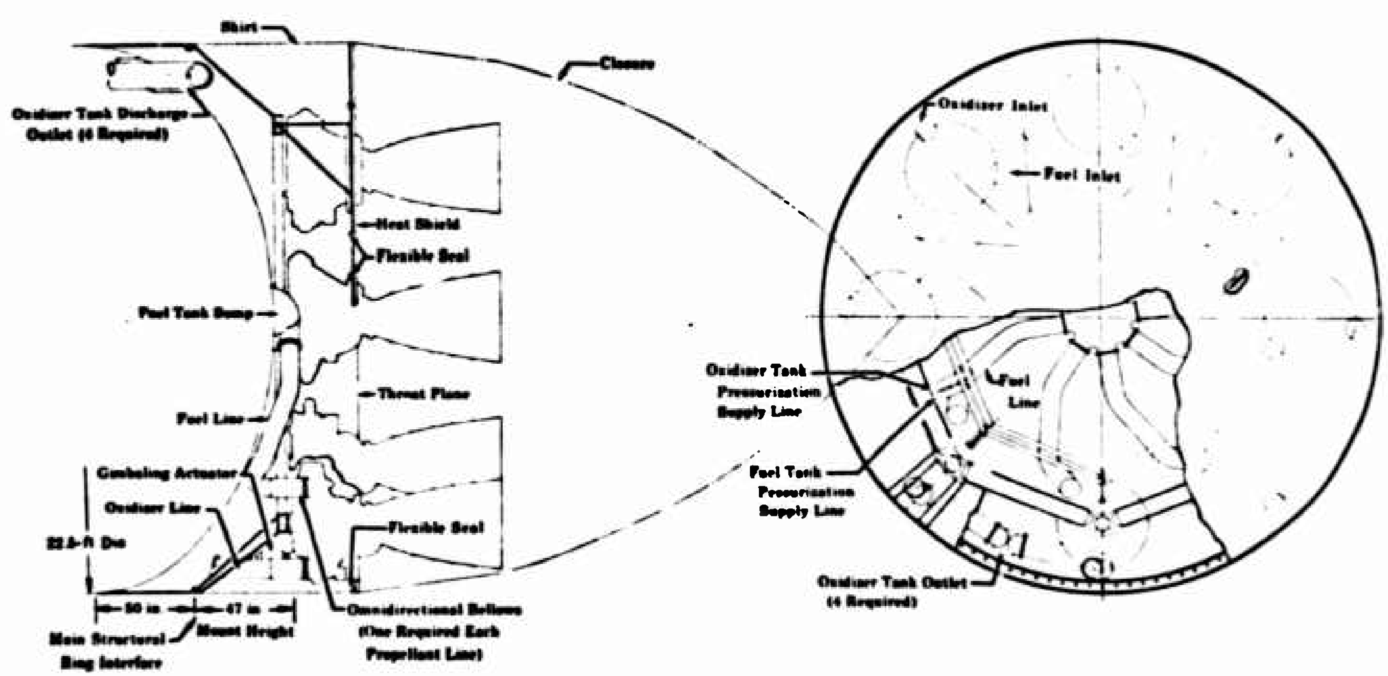
Figure 92. Case 3 Installation Layout



FD 23002

Figure 93. Case 4 Installation Layout

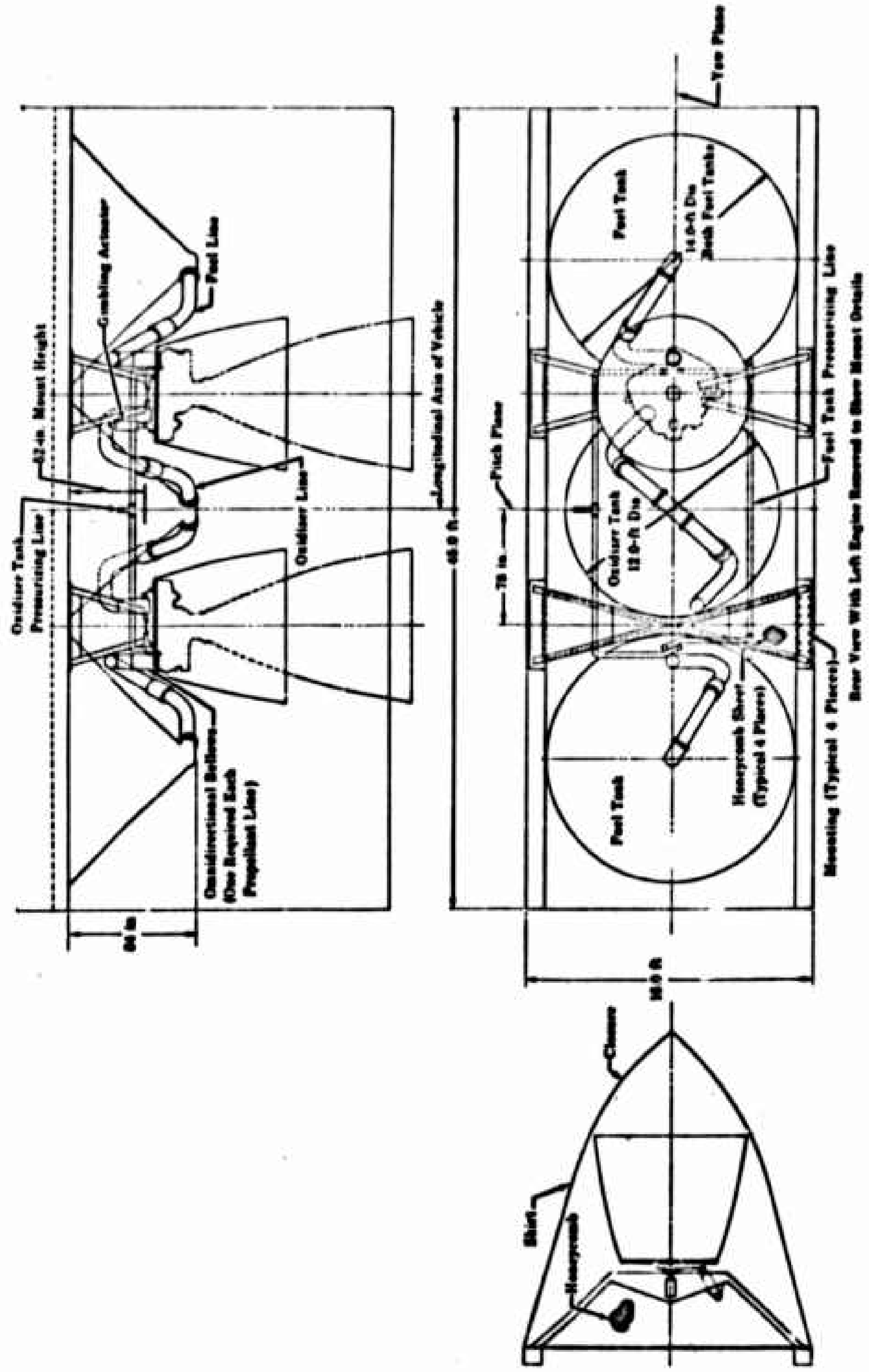


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Figure 94. Case 5 Installation Layout

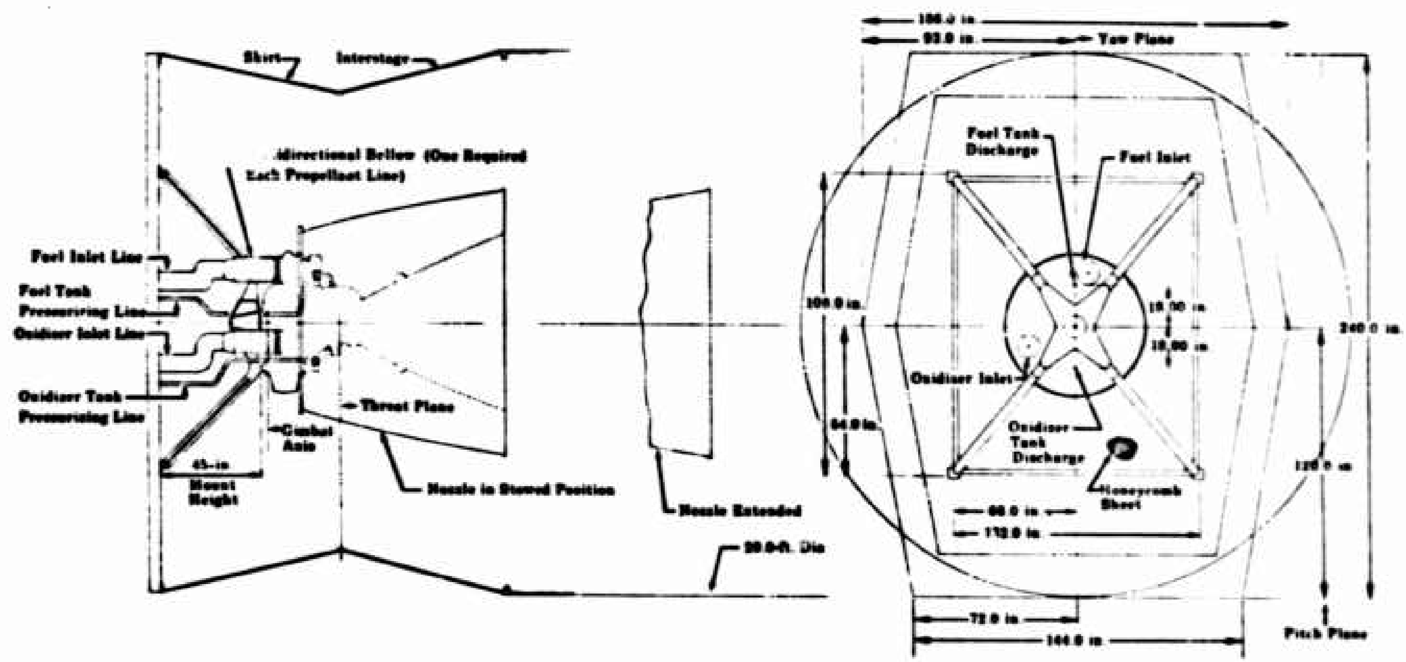


Figure 95. Case 6 Installation Layout

FD 22819

UNCLASSIFIED

SECTION V
SPECIAL STUDIES

A. NOZZLE CONCEPT EVALUATION

(U) A performance comparison study of the two-position lightweight nozzle concept relative to the fixed fully regenerative nozzle was conducted. The two-position lightweight nozzle produced higher performance than the fixed regenerative in all cases except Case 4. The difference in performance of the two concepts for Case 4 was small but favored the fixed regenerative nozzle. The gains in performance resulting from the lightweight two-position nozzle concept were further evaluated to establish the relative individual merits of the two-position feature and the lightweight feature. The performance gain in Cases 1, 3, and 5 was due mainly to the two-position nozzle while the lightweight feature was the most significant for Cases 2 and 6.

(U) The performance advantages of the two-position lightweight nozzle concept are due to:

1. Lighter engine weight - Nozzle is cooled by low-pressure hydrogen expelled overboard and, therefore, the cooling tubes do not have to be constructed to withstand the high internal working pressures associated with a regeneratively cooled nozzle.

2. Better sea level performance - The low area ratio portion of the nozzle can be used for lower stages at sea level and the high area ratio nozzle can be extended as the vehicle gains altitude.

3. Higher area ratios, and therefore higher specific impulse, can be packaged in a given interstage area for higher upper stage performance.

(U) Although the expelled hydrogen in a dump-cooled, lightweight nozzle has specific impulse levels equal to the main exhaust stream, a small engine performance loss results because this coolant flow extracts heat from the main exhaust stream that would normally be returned to the engine in a regenerative cooling cycle. A small loss also occurs because of a shift to higher injector mixture ratio when the dump coolant hydrogen is extracted from the cycle. The sum of these penalties, however, is less than 1 second in specific impulse for lower stage and less than 2 seconds for the upper stage area ratios. The nozzle translation mechanism weight penalties must be considered in comparison with fixed-nozzle engines. This is a trade-off of the length and weight sensitivity of the particular application.

(U) The Performance Index calculations provided a method to assess the effect on total vehicle performance of the lightweight, two-position nozzle, while including the effect of the slight performance degradation due to dump cooling. In addition, the individual aspects of the concepts can be evaluated.

(C) To obtain a direct comparison of the nozzle concepts, a common engine module sized to develop 250K vacuum thrust at a nozzle area ratio of 181 was used to establish performance for the concepts. The 181 expansion ratio is the common module size that was established for the fixed regeneratively cooled nozzle engine. The actual module flow rate used varied slightly between the regenerative and dump cooling concepts because of the small differences in specific impulse. Using the fixed module flow rate, the location, mixture ratio, and nozzle expansion ratio were optimized for the nozzle concepts.

(U) The results of this study are summarized in table XI, which gives the relative Performance Index for each case. The effect in all cases of using the two-position lightweight nozzle concept, relative to the fixed regenerative nozzle concept, is shown in the first column. To differentiate the gain due to the two-position and lightweight concepts, the effect of using a fixed lightweight nozzle is shown in the second column. All results are relative to the fixed regenerative nozzle.

(U) Table XI. Performance Index Comparison of Nozzle Concepts

	Percent Increase in W_x Relative to Fixed Regenerative Nozzles	
	Two-Position Lightweight	Fixed Lightweight
Lower Stages		
Case 1	1.0	0.2
Case 3	4.3	0.5
Case 4	-0.7	0.0
Upper Stages		
Case 2	1.1	0.8
Case 5	5.8	2.2
Case 6	5.8	6.9

(U) Figures 96 through 101 present the Performance Index (W_x) for the nozzle concepts as a function of the nozzle expansion ratio. All points on these curves are also optimized for mixture ratio (at each expansion ratio) and location. The use of the two-position lightweight nozzles increases the optimum expansion ratio for all cases, except Cases 2 and 6, which had optimized at the maximum area ratio for the fixed regenerative nozzle. The use of a fixed lightweight nozzle increases the optimum area ratio for Case 5.

(C) In the lower stage cases, the use of the two-position lightweight nozzle significantly increases the performance over the fixed regenerative nozzle except in Case 4. Case 3, which is a single-stage-to-orbit application, places a premium on both sea level and vacuum performance and, therefore, indicates the largest performance improvement of the lower stages. A comparison of the results in columns 1 and 2 indicate that the major increase in performance for Cases 1 and 3 is due to the advantages of the two-position nozzle concept, which allows both improved sea level performance and an increase in the optimum overall area ratio. Case 4 has a strong

diameter constraint that limits the obtainable expansion ratio, and, therefore does not show an advantage for the two-psoition concept. Further, the optimum expansion ratio for a fixed nozzle is approximately 35 and therefore there is no difference in performance for a regenerative or lightweight (dump cooling starts at an area ratio of 35) nozzle (figure 99).

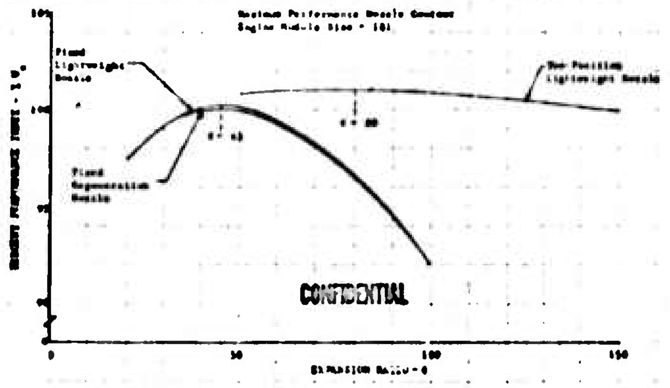


Figure 96. Percent Performance Index vs Expansion Ratio for Case 1, 250K Module — DF 60035

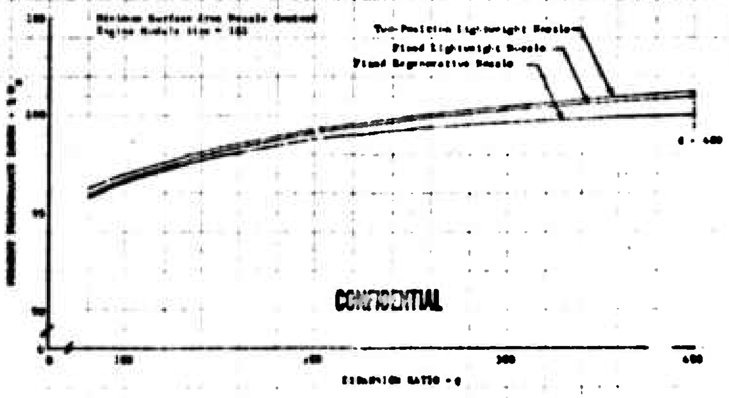


Figure 97. Percent Performance Index vs Expansion Ratio for Case 2, 250K Module — DF 60036

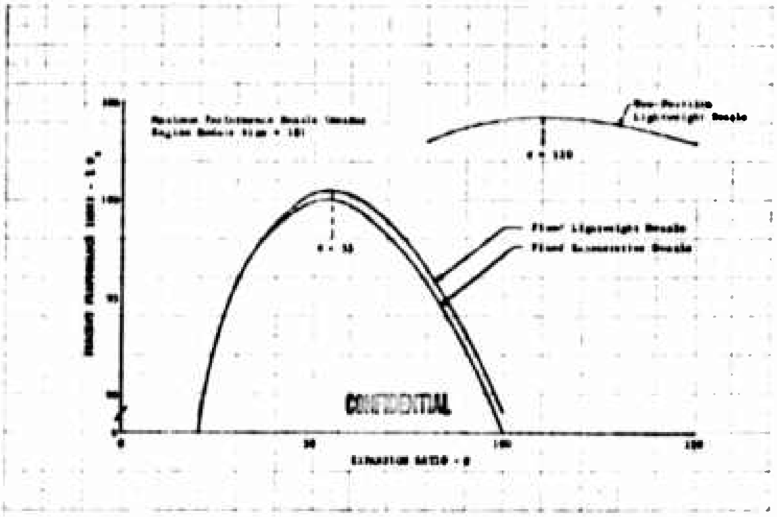


Figure 98. Percent Performance Index vs Expansion Ratio for Case 3, 250K Module

DF 60037

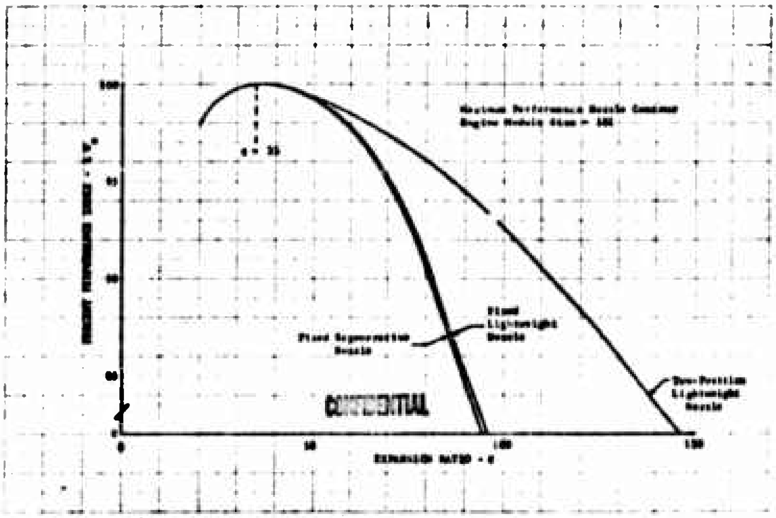


Figure 99. Percent Performance Index vs Expansion Ratio for Case 4, 250K Module

DF 60038

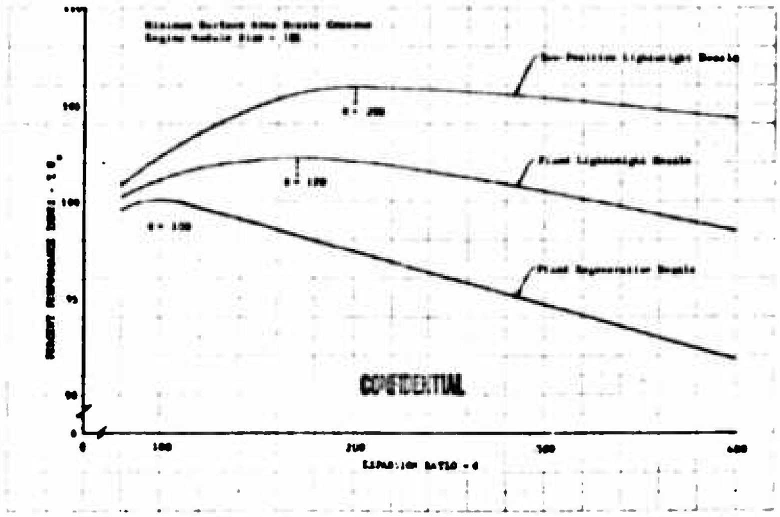


Figure 100. Percent Performance Index vs Expansion Ratio for Case 5, 250K Module — DF 60039

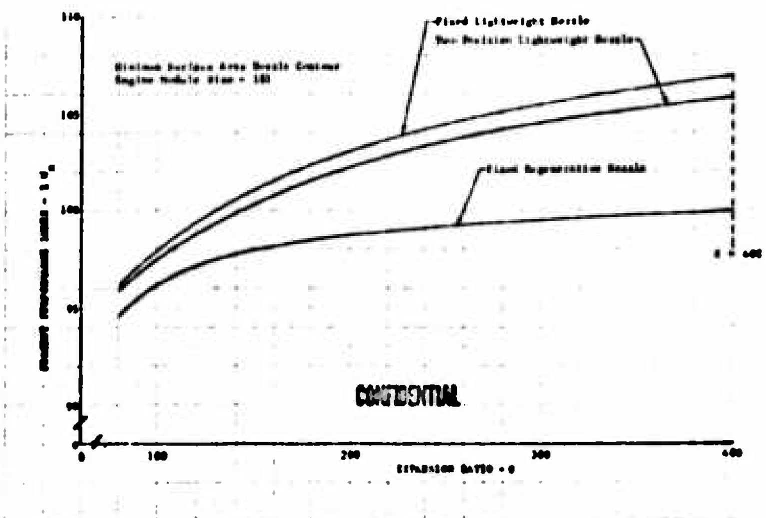


Figure 101. Percent Performance Index vs Expansion Ratio for Case 6, 250K Module — DF 60040

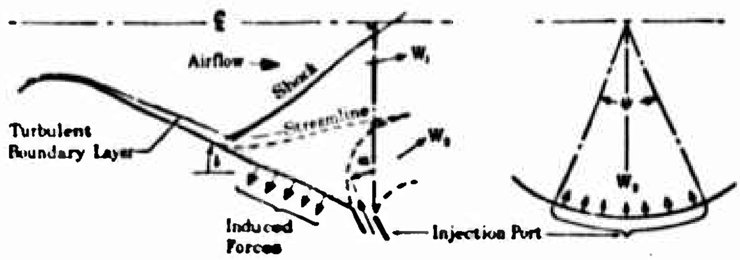
(U) For the upper stages, the maximum gain in performance for two-position lightweight nozzles was in Cases 5 and 6. The increase in performance was equivalent to a gain of approximately 11 seconds and 10 seconds in specific impulse for Cases 5 and 6, respectively. Case 2 yields less of a gain and this is due mainly to the lightweight feature. The results also indicate that the major advantage in Case 5 is due to the two-position concept and in Case 6 to the lightweight feature. Both Cases 5 and 6 are very sensitive to savings in engine weight because the W_x calculations includes a stage weight-growth constant that yields 3 lb increase in W_x for 1 lb decrease in engine weight. Case 5 has a strong length constraint (50 lb in W_x per inch of stowed length) due to the closure fairing area; whereas Cases 6 (although having an interstage area) has very little length constraint (1 lb of W_x per inch of stowed length). The reduced stowed length of the two-position concepts allows a significant increase in the nozzle expansion ratio in the length constrained Case 5, (figure 100). The length constraint in Case 6 is not strong enough to offset the additional weight of the two-position nozzle actuation mechanism. Case 6 optimizes at the highest expansion ratio for all nozzle concepts as shown in figure 101. Case 2 has a low length constraint (but greater than Case 6) of 2.7 lb in W_x per inch of stowed length and a one-for-one weight trade. This results in a modest performance improvement for the two-position lightweight nozzle with the major portion due to the lightweight feature.

B. THRUST VECTOR CONTROL COMPARISON ANALYSIS

(U) The objective of the Thrust Vector Control (TVC) system analysis was to select the optimum system for use in the application study cases. Selection between the competing systems, mechanical deflection and secondary injection thrust vector control (SITVC), was based on comparison of Performance Index (W_x) values for the six vehicle cases.

(U) The comparison indicated a distinct advantage of the mechanically gimbaled system over the secondary injection system. Therefore, the mechanical gimbal system was selected for all the application study cases.

(U) To compare the secondary injection system with a mechanically gimbaled system, a detailed study of SITVC was conducted that defined the system and outlined the requirements. The secondary flow injection technique depends on the side forces generated by a fluid injected into the nozzle exhaust stream to produce a resultant thrust vectoring force. The disturbance created results in an induced force operating on the nozzle wall mainly in the lateral direction. This induced side force acts in addition to the direct reaction from the fluid injection. Figure 102 schematically depicts the operation of a gaseous secondary injection TVC system. The side force is induced on the side of injection and there is little axial deflection of the main exhaust jet.



PARAMETERS AFFECTING PERFORMANCE OF TVC

PRIMARY NOZZLE	SECONDARY NOZZLE
Area Ratio,	Weight Flow, W_2/W_1
Pressure Ratio, P_R	Injection Angle, α
Size, Re_d	Slot Width, ϕ
Gas Properties, γ_1, $\mathcal{M}_1$, T_{T_1}	Mach Number, M_2
	Gas Properties, γ_2, $\mathcal{M}_2$, T_{T_2}

Figure 102. Flow Model Schematic of Secondary Injection for Thrust Vector Control FD 14058

(U) Injection efficiency, side force requirements, and duty cycle determine the injectant flow rates and total amount of injectant required for each application study case. The secondary injection systems were sized for a maximum side force-to-axial force ratio of 0.07. The average side force-to-axial force ratio used was 0.035 for the lower stages and 0.02 for the upper stages. A total side impulse-to-axial impulse ratio of 0.02 for the lower stages and 0.01 for the upper stages was used. These design requirements were specified in the Applications Study Package (Appendix I).

(U) The parameters and characteristics pertinent to fluid injector systems were examined to establish an appropriate basis for further system analysis. These parameters and results are discussed in the following paragraphs. References 1, 2, and 3 were used extensively in this review.

1. Type of Injectant

(U) Three general types of secondary injection systems were considered for the applications study cases. These were (1) hot gases such as combustion gases, (2) cool gas, and (3) reactive liquids. The specific fluid chosen from each group were oxygen-hydrogen combustion gases, room temperature hydrogen, and nitrogen tetroxide. Required injectant flow rates were determined for each fluid for the required side force-to-total force ratios. Figures 103 and 104 show the required maximum flow for a typical lower stage (Case 1) and upper stage (Case 2). A comparison of performance of N_2O_4 and hydrogen gas with that of combustion gases for the applications cases indicated that N_2O_4 and hydrogen gas can be eliminated as SITVC injectants on the basis of required injectant flow rate and total required injection pressures. Accordingly, only the hot combustion gas system was considered for design analysis.

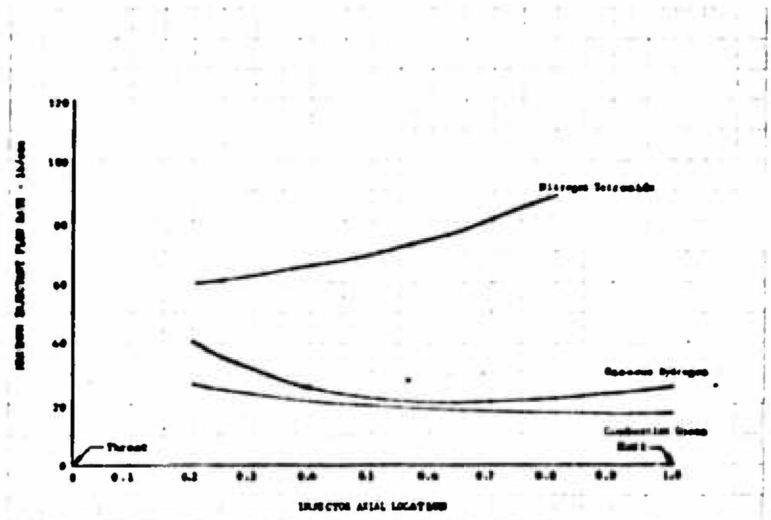


Figure 103. Maximum Injectant Flow Rate vs Injector Axial Location (Case 1) DF 57378

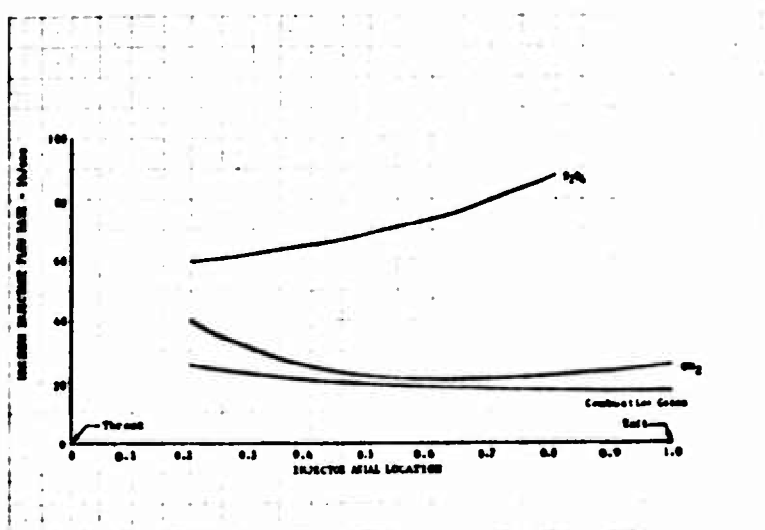


Figure 104. Maximum Injectant Flow Rate vs Injector Axial Location (Case 2) DF 57379

2. Injector Axial Location

(U) The optimum axial location varies for the type of injectant. Axial locations are expressed as X/L, the distance from the throat of the main nozzle to the TVC injector location, divided by the distance from the throat of the main nozzle to the exit plane of the main nozzle.

(U) Combustion gases exhibited the best performance when injected at one nozzle exit. (See figure 105.) The optimum X/L is 0.65 for room temperature hydrogen and 0.2 for nitrogen tetroxide.

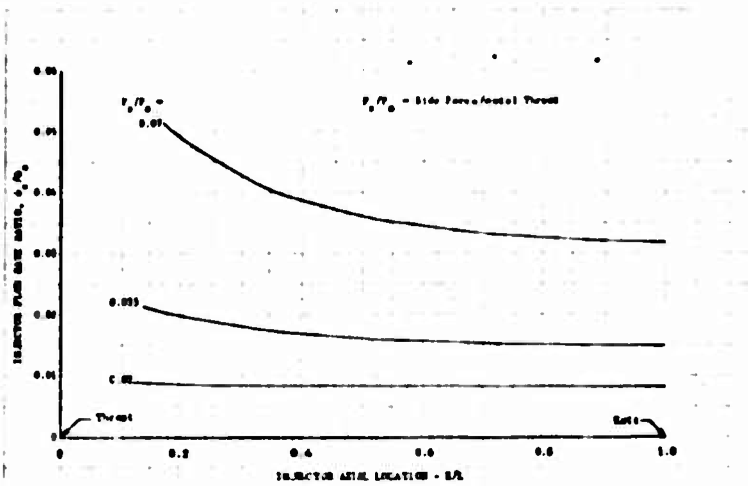


Figure 105. Weight Flow Ratio vs Axial Location (Combustion Gases) DF 57283

3. Injectant Mach Number

(U) The majority of available SITVC data are for injection of the fluid at Mach 1, therefore, the design Mach number for this study was assumed to be Mach 1.

4. Sector Angle

(U) The sector angle is the arc of the main nozzle circumference through which injection takes place. Available data showed that the optimum section angle was 45 degrees for combustion gares.

5. Temperature of Injectant

(U) Reference 3 presents a parameter that can be used to correlate gaseous data with different molecular weights and temperatures

$$S = \frac{T_{T_2}}{T_{T_1}} \left(\frac{\gamma_1}{\gamma_2}\right)\left(\frac{\gamma_2 + 1}{\gamma_1 + 1}\right)\left(\frac{\mathcal{M}_1}{\mathcal{M}_2}\right)$$

This parameter can be used to estimate performance for different injection temperatures by multiplying by the ratio of similarity parameters. The next phase of the secondary injection analysis included the performance evaluation and integration of the system with the engine and vehicles being considered in the application study.

(U) Two types of secondary injection systems using combustion gases were considered for preliminary design. The first system used hot gases extracted from the main engine cycle (tapoff system) and the second system used hot gases from a separate gas generator (separate system).

(U) Design analysis to obtain the weight of the equipment associated with each system was conducted. The analysis was based on the flow rate and total injectant requirements of each application case.

(U) The tapoff system, which extracts hot gases from the main engine cycles, was conceptually designed to extract hot gases from below the turbines. The components for the tapoff system include supply lines, control system, manifold, thrust nozzle stiffening requirements, and four separately controlled gas injecting manifolds centered at 90-degree intervals with each extending over an arc of 45 degrees. In addition, the system weight includes estimates for these components and the total injected gas weight.

(U) In the separate secondary injection TVC system, the gas for the injection into the main engine nozzle is obtained from a gas generator. Propellants (H_2 and O_2) for the gas generator are supplied from separate pressurized tanks. The liquid oxygen and hydrogen tanks were sized for a propellant mixture ratio of 1.30. The tanks were pressurized by helium stored under high pressure at a temperature of 30°R. A weight optimization was made for the initial helium tank storage pressure. In addition, weight analysis includes control ignition system supply lines, injecting manifolds and nozzle stiffening. Arrangement of the system components was assumed to cause no increase in vehicle length.

(U) The installation weight for the secondary injection TVC schemes was obtained for all cases and is shown in table XII relative to the mechanical gimbal installation weights.

(U) Table XII. Installation and Propellant Weight Changes for Secondary Injection Systems

Case	Installation Weight* Tapoff, lb	Installation Weight* Separate System, lb	Propellant Tapoff lb	Propellant Separate System, lb
1	+1035	+6050	4500	4500
2	+ 465	+1225	700	700
3	+1035	+8900	6150	6150
4	+1640	+9680	7200	7200
5	+ 930	+2450	1400	1400
6	+ 460	+1225	725	725

*Relative to mechanical gimbal system.

(U) A Performance Index (W_x) comparison of each system was compared to mechanical gimbal baseline values. The result of using either hot-gas secondary system was a significant reduction in W_x for all six ADP vehicle cases. Table XIII provides a summary of the W_x results.

(U) Table XIII. Performance Index Comparison of Hot-Gas Secondary Injection to Mechanical Gimbal (% Difference)

	Tapoff	Separate System
Case 1	-1.91	-3.81
Case 2	-1.57	-2.51
Case 3	-9.75	-25.0
Case 4	-3.25	-9.6
Case 5	-5.9	-9.2
Case 6	-6.25	-9.4

(U) The separate system requires its own pressurant and propellant tanks and gas generator so that no impulse or tankage mixture ratio penalty need be assessed. The W_x values were calculated at the previously established baseline optimum mixture ratio and expansion ratios.

(U) The weight of the ancillary tanks and propellants required for the external secondary injection system was considered; however, the results in table XIII are based on the assumption that tank placement would not interfere with optimum engine location (i.e., the tanks were considered to be stored above the vehicle-engine interface). Case 1 was examined to investigate the magnitude of penalty incurred by storing the tanks below the vehicle-engine interface. Tank diameters for the hydrogen, oxygen, and helium are 5.6, 2.4, and 4.4 feet, respectively, for each engine. The engine spacing and height were varied so that the tanks would fit symmetrically around the vehicle. This tank arrangement further decreases the Case 1 performance index to 5.34% as compared with a decrease of 3.81% for the noninterference arrangement (table XIII).

(U) The tapoff cycle system must be penalized for extracting flow from the engine cycle when W_x performance is calculated. This can be done in either of two methods; either by using an engine impulse and thrust loss caused by flow extraction or by compensating the engine by utilizing a different mixture ratio from the main propellant tanks. The results in table XIII are based on shifting the tank mixture ratio to compensate for flow extraction. To check the validity of using this method of analysis, the W_x for Case 5 was calculated using both methods; the engine performances shift method gave 5.95% change as compared to 5.9% for the mixture ratio change.

(U) Based on the results of table XIII it was concluded that the mechanically gimbaled system gives the best performance for all cases.

C. PROGRAMMED MIXTURE RATIO

(U) A study was conducted to determine the effects and possible Performance Index gains by using a programmed mixture ratio for lower stage applications. This technique uses high mixture ratio operation during the early portion of the vehicle flight to reduce vehicle gross weight more rapidly at a sacrifice in specific impulse; subsequently, module operation reverts to the previously determined optimum mixture ratio (maximum impulse) for the remainder of the flight. The anticipated performance improvement results from the increased vehicle acceleration during the early acceleration sensitive portion of the trajectory. The higher acceleration results from the higher propellant flow rates at high mixture ratios. The lower specific impulse at high mixture ratios is countered by operation at low mixture ratios and higher specific impulse during the latter, less thrust sensitive portion of the boost trajectory.

(U) Two specific engine operating modes were investigated:

1. Using the engine module over-thrust capability for all but the extreme high and low mixture ratio points

2. Normal operation (i.e., constant thrust over the mixture ratio range).

(U) These thrust profiles are shown in figure 106. The upper curve (type 1) shows the available thrust, which is defined from the engine operational limit of maximum design fuel pump speed. The lower or constant curve (type 2) is the normal operating mode.

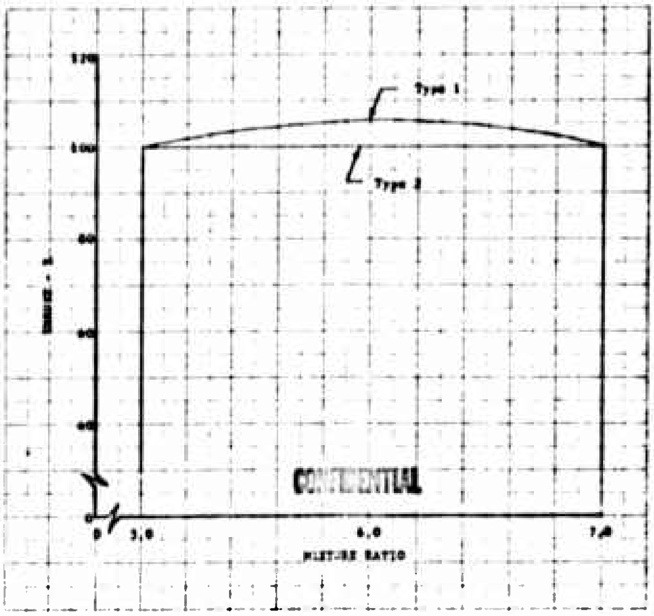


Figure 106. Thrust vs Mixture Ratio DF 57557

(U) The effect of programmed or varying mixture ratio on the overall Performance Index was established by optimizing the level of the high mixture ratio and the time (vehicle velocity increment) when the mixture ratio is reverted back to a low value. As discussed previously and outlined in Appendix I, vehicle calculations are made at velocity increment steps of 200 fps over the complete velocity increment (ΔV) range. The study was done by programming the vehicle-engine to operate at a series of higher than average mixture ratios at the beginning of the flight. The velocity (ΔV) at which the system was transferred back to a lower mixture ratio was also varied. This is shown in figure 107. For the mission calculation, the mixture ratio is set at level (A) to a ΔV value of (1) at which point the mixture ratio is changed to level (B). A Performance Index (W_x) value is obtained for this mission profile. This procedure is repeated for several initial portions of ΔV values (2), (3), (4), etc. This is then repeated for other initial or primary mixture ratios such as (C). The various obtained values of the Performance Index (W_x) are plotted in figure 108 for Case 1. This curve gives the Performance Index at various primary mixture ratios as a function of the ΔV portion of the mission where the initial mixture ratio was returned to the lower mixture ratio. The lines of primary mixture ratio represent a locus of performance points using the given initial mixture ratio to successive higher ΔV values. These data can be cross-plotted to yield the improved W_x along with the optimum primary mixture ratio and optimum transition ΔV valves. These curves are restricted to a given engine area ratio. This complete procedure was repeated for various engine area ratios to reoptimize the engine area ratio for the programmed mixture ratio operation.

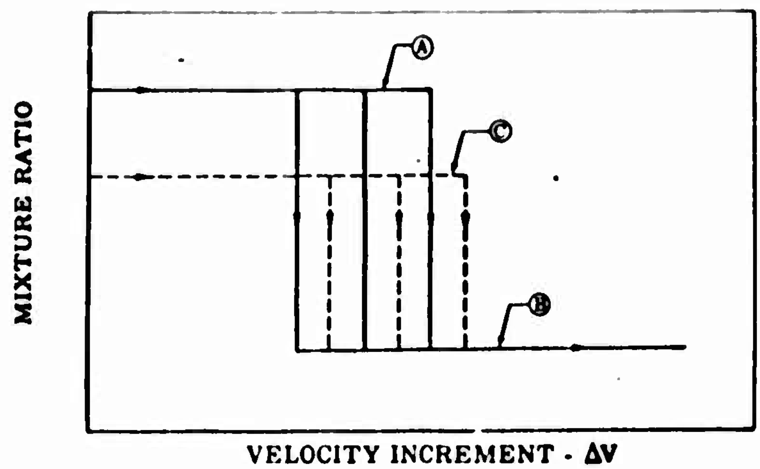


Figure 107. Mixture Ratio vs Velocity Increment FD 21997

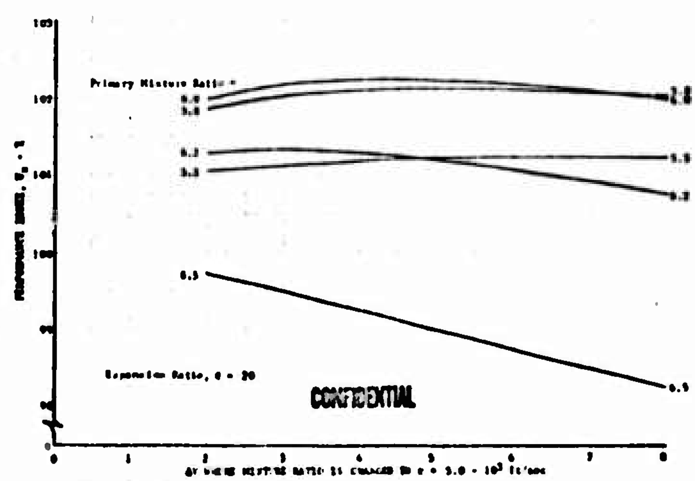


Figure 108. Programmed Mixture Ratio With Variable Thrust (Case 1), Type 1 DF 57380

(U) The final results for each of the lower stage cases using the increased thrust profile is shown in figures 109, 110, and 111. These results indicate that the engine available thrust and higher propellant flow rate at high mixture ratios did produce trajectory improvements sufficient to outweigh the losses resulting from the lower specific impulse during high mixture ratio operation. The net increase in Performance Index (W_x) is shown in table XIV.

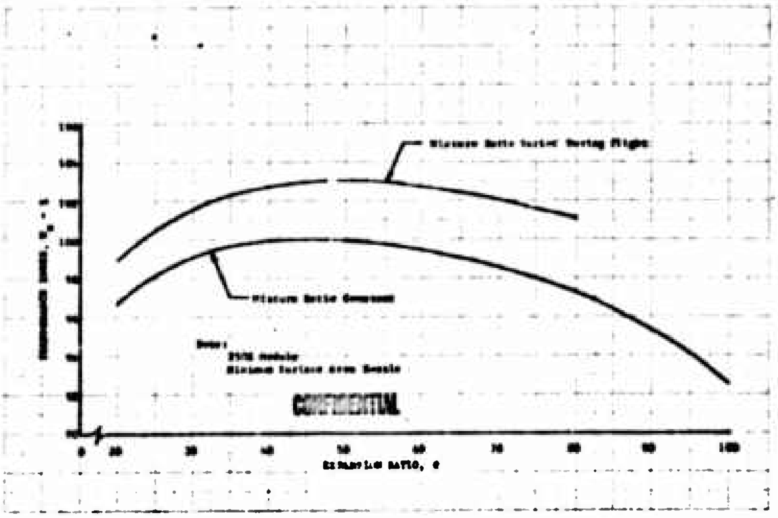


Figure 109. Programmed Mixture Ratio for Case 1, Type 1 DF 57381

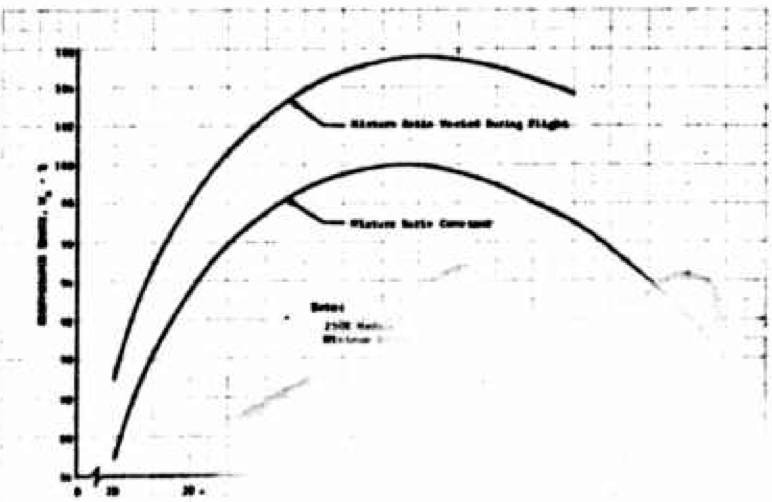


Figure 110. Progr
Type 1

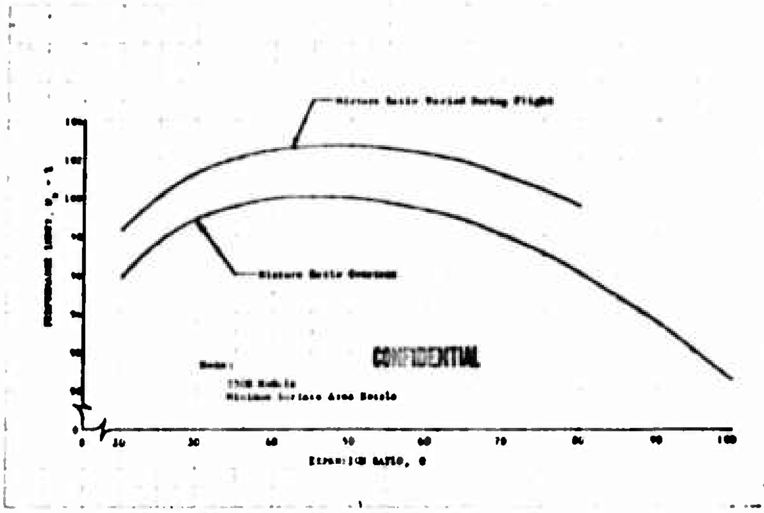


Figure 111. Programmed Mixture Ratio for Case 4, Type 1 DF 57383

(U) Table XIV. Performance Improvement Using Programmed Mixture Ratio

Case	Increase in W_x, %
1	3
3	5.5
4	2.6

(C) The effects of programmed mixture ratio was also evaluated with constant thrust over the mixture ratio range to separate the effects of vehicle mass change rate from the influence of thrust increase. Analysis indicated reduced Performance Index values for all cases and for all mixture ratio variations. Typical results obtained are shown in figure 112 and all performance values are below the constant, but optimum, mixture ratio evaluation. Similar analysis for Cases 3 and 4 yield the same conclusions as for Case 1. The evaluation of various area ratios were limited due to the negative results; however, area ratios of 40 and 60 were checked and gave the same conclusion. Therefore, with the constant thrust operation, the improvements due to the higher acceleration (reduced gravity and drag losses) during high mixture ratio operation did not outweigh the losses due to the lower specific impulse.

(U) The conclusion of the investigation of programmed mixture ratio was that programmed mixture ratio offers significant improvements in performance only when associated with using the available increased engine thrust.

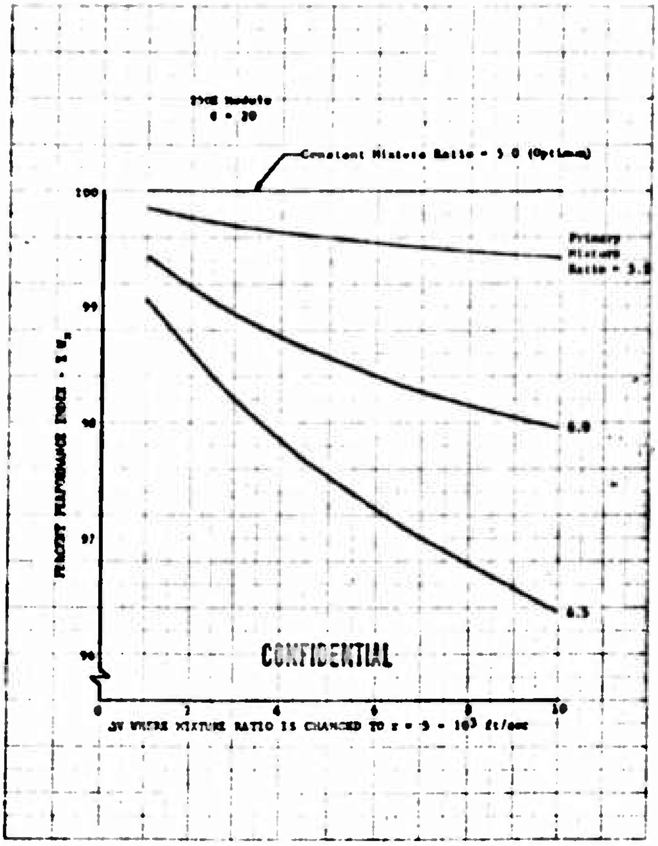


Figure 112. Programmed Mixture Ratio for Constant Thrust, Type 2 DF 52884

D. SENSITIVITY PARTIALS

(U) The sensitivity of the Performance Index to various engine parameters was determined. These factors were determined by varying the parameters individually for each application case. The results are listed in table XV, in terms of trade factor as W_x in pounds per unit change in the parameter.

(U) Table XV. Performance Index Sensitivity to Engine Parameters

Case	Engine Weight, lb/lb	Engine Stowed Length, lb/in.	Engine Impulse, lb/sec (At Constant Flowrate)	Engine Flow Rate, lb/lb/sec (At Constant Impulse)	Engine Diameter, lb/in.
1	-5	0	947	320	—
2	-1	-2.7	218	11	—
3	-5	0	533	111	—
4	-20	-20	1588	506	1
5	-6	-50	415	22	40
6	-3	-1.0	226	11	—

E. ALTERNATIVE MODULE SIZING TECHNIQUES

(U) A requirement of the study was that a common engine module be selected and used in all six vehicle cases. The exhaust nozzles used with this common module can vary in expansion ratio or configuration from case to case to produce optimum stage performance. With the high-pressure staged combustion engine concept, different bell nozzles can be used with the same basic engine module. It was determined that the application cases in this study were sensitive to propellant flow rate and, therefore, to the technique used to size the basic engine module. Because the module is defined by vacuum thrust, increasing the basic module size to higher propellant flow rates produced higher Performance Index in the lower stages. The effect of the flow rate is shown in Section II.

(C) Because of the flow rate effect, it was decided to investigate a number of optimization techniques before the final analysis. The results of this study in terms of a normalized summation of W_x values are shown in figure 3. Table XVI presents a breakdown of the individual stage performance for each of the six cases. The recommended technique (and used for the final W_x values in this report) is shown on the second bar. A normalized summation of all six vehicle cases was used to establish the expansion ratio for 250K vacuum thrust and, thus, common module size. No engine exceeds 250K vacuum thrust; however, higher performance could be obtained with this technique if Cases 2 and 6 were allowed to go to the stage matched optimum of 400. The summation percentage would increase from 98.2 to 98.5.

(C) Table XVI. Summary of Alternative Module Sizing

	Case 1	Case 2	Case 3	Case 4	Case 5	Case 6
1. All Cases, F_{vac} = 250K Individual Power Packages and Nozzles	100.0 ϵ = 50	100.0 ϵ = 400	100.0 ϵ = 105	100.0 ϵ = 35	100.0 ϵ = 200	100.0 ϵ = 400
2. Common Power Package Stage Matched Nozzles ($F_{vac} \leq$ 250K) F_{vac} = 250K at ϵ = 250)	97.9 ϵ = 75	98.8 ϵ = 250	98.3 ϵ = 110	96.8 ϵ = 35	99.9 ϵ = 205	97.8 ϵ = 250
3. Common Power Package Stage Matched Nozzles ($F_{vac} \leq$ 250K) F_{vac} = 250K at ϵ = 400)	97.1 ϵ = 75	100.0 ϵ = 400	96.6 ϵ = 108	96.1 ϵ = 35	99.8 ϵ = 220	100.0 ϵ = 400
4. Common Power Package F_{vac} = 250K at ϵ = 400 ($F_{vac} \leq$ 250K: ϵ = 400 in Cases 2, 5, 6)	97.1 ϵ = 75	100.0 ϵ = 400	96.6 ϵ = 108	96.1 ϵ = 35	98.2 ϵ = 400	100.0 ϵ = 400
5. Common Power Package Stage Matched Nozzles ($F_{vac} \leq$ 250K) F_{vac} = 250K at ϵ = 181	98.2 ϵ = 80	97.8 ϵ = 181	99.3 ϵ = 110	97.5 ϵ = 35	99.7 ϵ = 181	95.6 ϵ = 181
6. Common Power Package Single Two-Position Lightweight Nozzle F_{vac} = 250K at ϵ = 95	98.8 ϵ = 95	95.9 ϵ = 95	99.5 ϵ = 95	92.6 ϵ = 95	96.2 ϵ = 95	91.8 ϵ = 95
7. Common Power Package Common Fixed Lightweight Nozzle F_{vac} = 250K at ϵ = 60	98.7 ϵ = 60	93.9 ϵ = 60	96.8 ϵ = 60	97.7 ϵ = 60	93.3 ϵ = 60	88.0 ϵ = 60
8. Common Power Package Common Fixed Regenerative Nozzle F_{vac} = 250K at ϵ = 58	98.7 ϵ = 58	93.6 ϵ = 58	96.8 ϵ = 58	97.6 ϵ = 58	93.2 ϵ = 58	87.4 ϵ = 58

(C) The other bars on figure 3 show the effect of common engine propellant flow and/or different ground rules. Bar No. 3 shows performance where a common engine size of 400 was selected. This was the optimum module size using the stated Applications Package optimization technique except that all engine expansion ratios were matched to stage requirements; however, no stage matched engine produces thrust above 250K vacuum thrust. (The summed W_x performance shown on Bar No. 3 was for two-position lightweight nozzles as were all other bars except as noted on Bar No. 7 and 8.) Bar No. 4 is the same as Bar No. 3 except that all upper stage engines used the same expansion ratio. Bar No. 4 thus exactly uses the optimization and module sizing technique (for two-position nozzle engines) as outlined in the Applications Package. Bar No. 5 indicates performance where a common engine size of 181 was selected. This was the optimum size for fixed regenerative cooled engines using the stated Applications Package optimization technique. Bar No. 6 shows W_x performance where the same expansion ratio two-position nozzle and the same power package was used in all six vehicle applications. Bar No. 7 is the same as Bar No. 6 except that a fixed lightweight nozzle engine is used in all cases. Bar No. 8 shows performance using a common power package and common fixed regenerative nozzle in all cases.

(U) Results of some of the more significant alternative optimization techniques are discussed in the following paragraphs.

1. Constant 250K Vacuum Thrust (Bar No. 1)

(U) Each application case was parametrically optimized with 250K vacuum thrust level at all points. The resulting Performance Index values are the highest possible within the ground rule that no stage matched engine can exceed 250K vacuum thrust, and provide a standard for rating other module sizing techniques. The optimization procedure was the same as discussed in Section IV. It should be noted that the thrust will be maintained constant with differing expansion ratios and a slight variation in module flow rate and power package size will occur. The parametric engine data used are in Appendix II.

a. Case 1 - Lower Stage Expendable

(U) Performance Index as a function of expansion ratio is shown in figure 113. With the constant thrust ground rule, the lower stages tend to optimize at lower expansion ratios, because at lower expansion ratios the overall propellant flow is increased to produce a given vacuum thrust. This increased propellant flow level raises sea level thrust which in turn provides better lower stage performance because of reduced gravity and drag losses. Case 1 expansion ratio is thus driven to such a level that a fixed nozzle engine has approximately the same performance as a two-position nozzle engine. This is also shown in figure 113. This is a somewhat artificial optimization stiuation; if the engine module size were fixed, the optimum expansion ratio would again increase to the values shown on the common module size study with an increase in vehicle performance with a two-position nozzle.

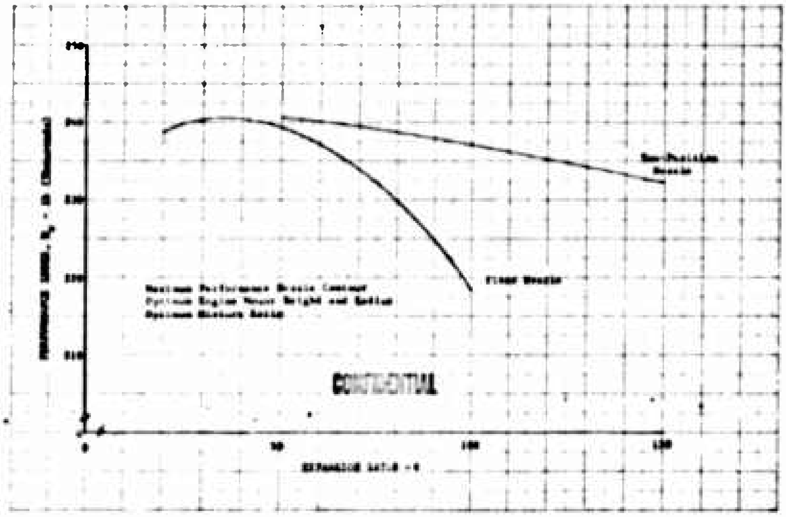


Figure 113. Performance Index vs Expansion Ratio for Case 1, 250K Module DF 57647

b. Case 2 - Upper Stage Expendable

(C) Performance Index as a function of expansion ratio is shown in figure 114. As noted with other module sizing ground rules, this case tends to optimize at the highest possible expansion ratio. A maximum expansion ratio of 400 was used in all of the application studies and Case 2 consistently shows increasing performance up to this limit with all techniques of optimization.

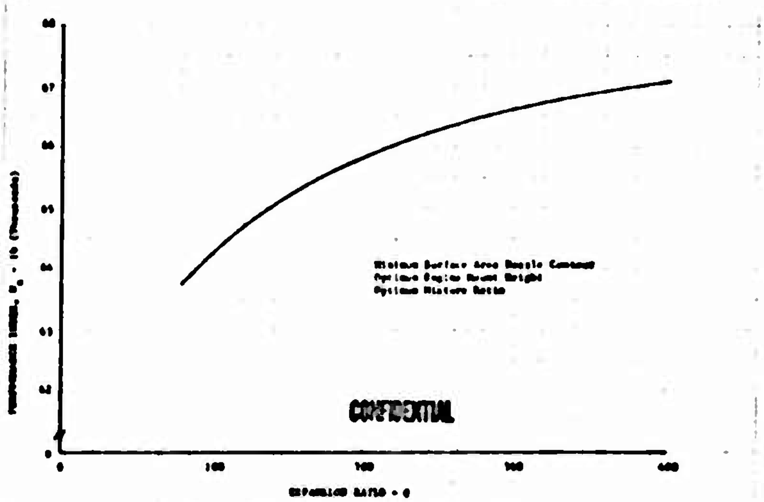


Figure 114. Performance Index vs Expansion Ratio for Case 2, 250K Module DF 57648

c. Case 3 - Single Stage to Orbit

(C) Performance Index as a function of expansion ratio is shown in figure 115. The optimum expansion ratio is 105, which is only a slight reduction from the optimum value obtained from the common module size study.

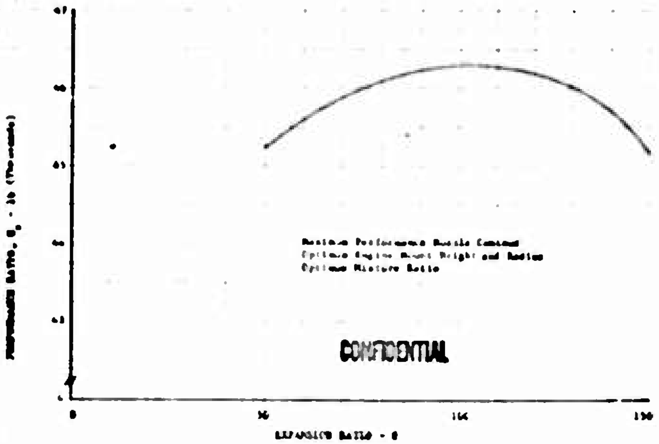


Figure 115. Performance Index vs Expansion Ratio for Case 3, 250K Module DF 57649

d. Case 4 - Recoverable Lower Stage

(C) Performance Index as a function of expansion ratio is shown in figure 116. The optimum expansion ratio is 35 (fixed nozzle), which is the same as that for the common module size. No change occurred because diameter restraints of the Case 4 vehicle forced the common module expansion ratio down to almost the lowest feasible value.

e. Case 5 - Recoverable Upper Stage, Pick-a-Back

(C) Performance Index as a function of expansion ratio is shown in figure 117. The optimum expansion ratio is 200 which is the same as that for the common module case.

f. Case 6 - Recoverable Upper Stage, Tandem

(C) Performance Index as a function of expansion ratio is shown in figure 118. As with Case 2, this case tends to show highest performance at the highest possible expansion ratio, which was 400 for all cases in the applications study.

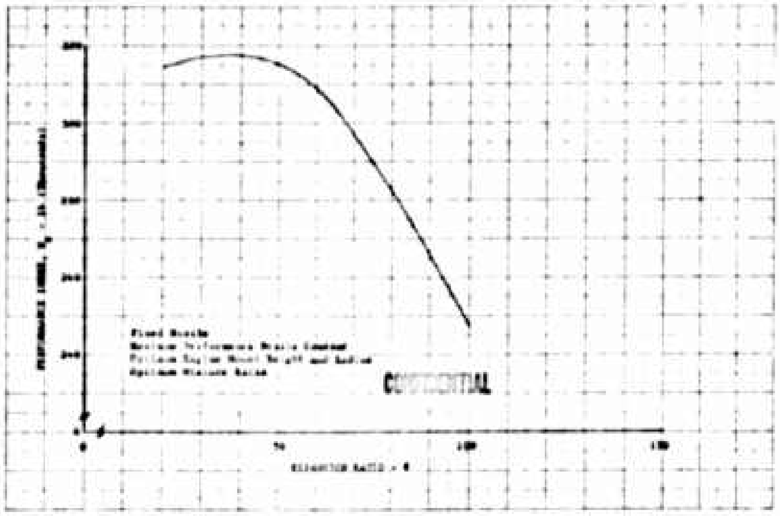


Figure 116. Performance Index vs Expansion Ratio for Case 4, 250K Module DF 57650

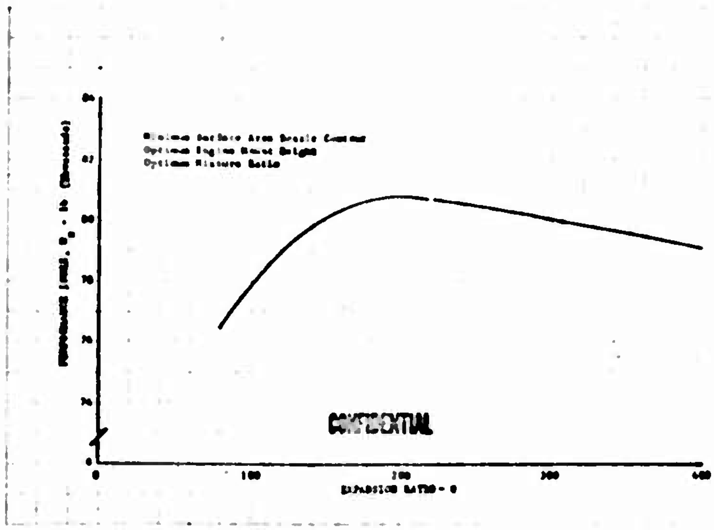


Figure 117. Performance Index vs Expansion Ratio for Case 5, 250K Module DF 57651

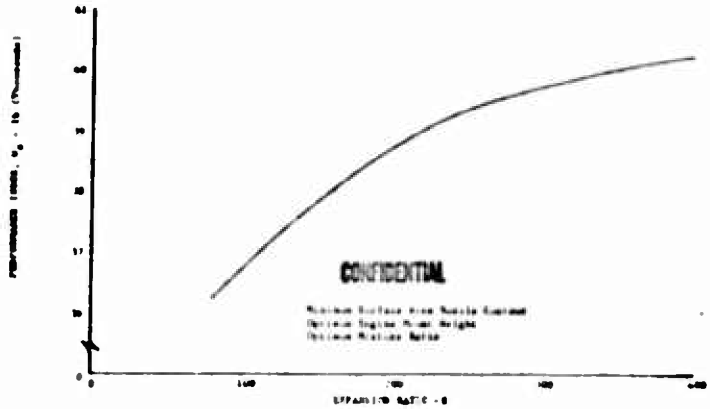


Figure 118. Performance Index vs Expansion Ratio for Case 6, 250K Module DF 57652

(U) A complete summary for all of the constant 250K vacuum thrust cases is shown in table XVII. For comparison, the results of 250K common module study are shown in table XVIII. The common module performance values are very close to the constant thrust values (which represent the highest case performance). This indicates that there is only minimal performance penalty associated with the use of a common engine module for all presently conceived vehicle applications.

2. Stage Matching Cases 2 and 6

(C) As noted in previous sections, Cases 2 and 6 individually optimize with expansion ratios of 400 or higher. A common module defined from the 250K study with an expansion ratio of 250 and stage matching these cases would yield more than 250K thrust if used with a 400 expansion ratio nozzle. However, it is of interest to consider the effects and possible gains for this procedure. The expansion ratio optimization curves are provided in figures 46 and 54 for Cases 2 and 6, respectively. A complete summary of the performance of these cases is given in table XIX. Performance Index of Case 2 would be improved 1.3% and Case 6 2.2% by increasing the expansion ratio from 250 of the common module to an expansion ratio of 400.

(C) Cases 2 and 6 for the 350K study were also optimized at the higher expansion ratio. The effect of increasing the expansion ratio of 300 for the common module to 400 is shown in figures 71 and 79 and is 0.7% improvement for Case 2 and 1.2% for Case 6.

(C) Table XVII. Performance Index (W_x)

	Case 1	Case 2	Case 3	Case 4	Case 5	Case 6
Performance Index (W_x)	240,700	67,100	46,300	317,000	80,850	40,200
Expansion Ratio	50	400	105	35	200	400
Mixture Ratio	5.5	6.0	5.75	5.75	6.44	6.72

Each Case With 250K Vacuum Thrust Engine(s)

(U) Table XVIII. Comparison of Individual Engines and Common Module

Case	Performance Index (W_x), lb Individual 250K Engines	Common Module
1	240,700	235,600
2	67,100	66,350
3	46,300	45,180
4	317,000	306,100
5	80,850	80,850
6	40,200	39,350

(C) Table XIX. Stage Matching of Cases 2 and 6 Common Module (ϵ_M = 250)

	Case 2		Case 6	
	Common Module	Stage Matching	Common Module	Stage Matching
Performance Index (W_x)	66,350	67,150	39,350	40,200
Vacuum Thrust (lb)	250,000	252,400	250,000	252,900
Engine Module Size	250	250	250	250
Expansion Ratio,	250	400	250	400
Mixture Ratio, r	5.9	6.03	6.7	6.76

3. Single Upper Stage Exhaust Nozzle

(U) The optimization technique used to generate the data presented in this report was to sum the performance of all six vehicle cases, choose a common module size, and optimize expansion ratio for each case for highest case performance. The comparative performance obtained using this technique was shown on Bar No. 2, figure 3. This technique produces almost identical summed performance to the original technique outlined in the applications package and shown as Bar No. 4 in figure 3. Here the three upper stages are used to size the common module. Using this common module, the exhaust nozzle expansion ratio is matched to each lower stage requirement, but used as a single nozzle expansion ratio engine in the upper stages. The main reason for selecting the technique used in this

report (Bar No. 2, figure 3) was that it balanced upper and lower stage requirements better and also that lower values of upper stage expansion ratio could be used. It is of interest to evaluate upper stage performance, however, using the same nozzle expansion ratio in all three stages. This is done in figure 119 for the 250K study and figure 120 for the 350K study.

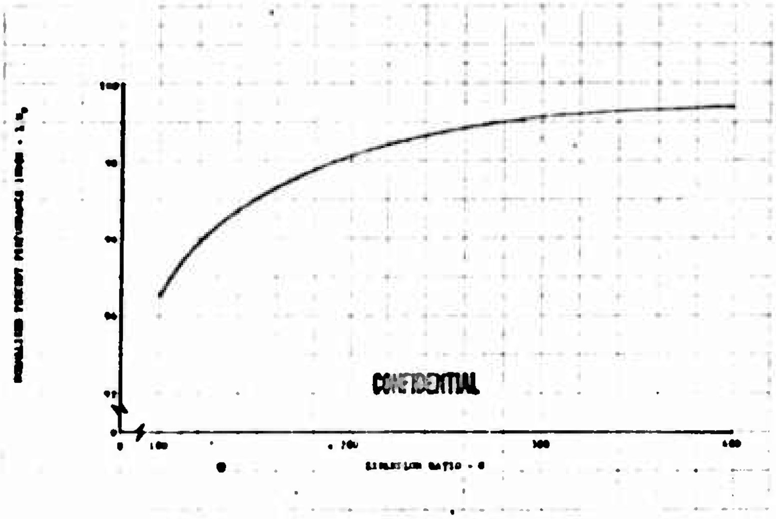


Figure 119. Normalized Percent Performance Index vs Expansion Ratio for Cases 2, 5, and 6, 250K Module DF 57653

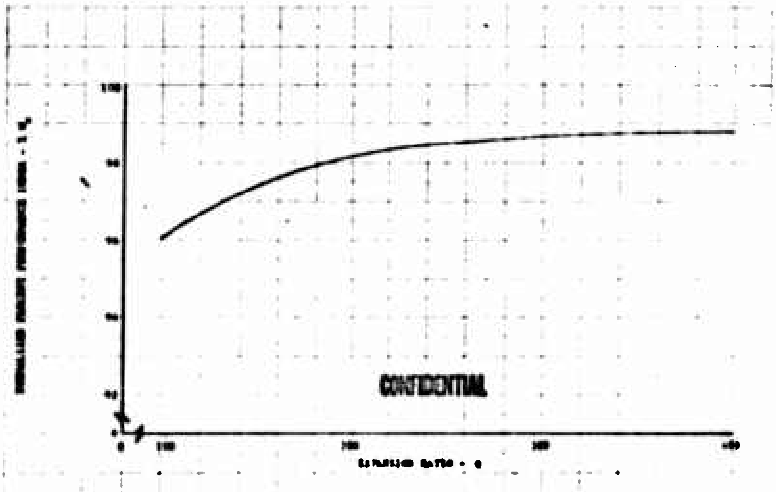


Figure 120. Normalized Percent Performance Index vs Expansion Ratio for Cases 2, 5, and 6, 350K Module DF 57656

(U) These curves show summed performance (normalized to 100%) of the three upper stages versus a common expansion ratio. This expansion ratio is the common ratio of all three stages at that particular point on the curve. The 100% level is defined as the summed performance of all three stages each using its optimum (and possible different) expansion ratio; this has previously been referred to as stage matched performance.

(C) The optimum common expansion ratio for the 250K upper stage module is 400 and is 0.6% lower than stage matched performance. The optimum common expansion ratio for the 350K upper stage module is 400 and is approximately 0.1% lower than stage matched performance.

SECTION VI
INSTALLATION STUDIES

(U) The installation weight term (W_e) in the Performance Index (W_X) calculation includes the weight of specified engine associated components. The installation analysis determined the installed weight of (1) thrust structure, (2) propellant feed lines, (3) heat shields, (4) pressurization systems, (5) failure detection systems, and (6) thrust vector control for each of the 12 cases studied. The weights determined by this analysis and a substantiating discussion for each case is provided in this section. To permit optimization of the engine installation, the weight analysis was performed parametrically as a function of the engine mounting location for the applicable items. Engine mounting locations were then optimized from the Performance Index (W_X) calculations. The installation dimensional nomenclature is shown in figure 89.

A. THRUST STRUCTURE

(U) Thrust structures are required to transmit the engine(s) thrust to the specified vehicle thrust loading mount in each of the cases.

(U) In this study the following ground rules were used:

1. Thrust structures designed to a spring constant of no less than 2.0×10^6 lb/in.

2. Structures independent of the vehicle mount for support (i.e., no radial restraint required of the vehicle attachment).

3. Equally distributed axial loads to be transmitted to vehicle thrust mounting.

4. Stress limit set at 66.7% (1.50 factor of safety) of the materials ultimate strength.

5. The five-engine installation of Cases 1 and 3 mounted in a cruciform arrangement (i.e., one on vehicle centerline). All other lower stage cases have engines mounted in a ring.

6. Engine thrust plus 10% used for thrust structure stress calculation.

1. 250K Module

(U) The number of engines and the supplied vehicle mount for each of the six cases using the 250K engine module is presented in the following list.

Case	Number of Engines	Supplied Vehicle Mount
1	5	276-in. diameter ring
2	1	204-in. diameter ring
3	5	312-in. diameter ring
4	8	270-in. diameter ring
5	2	Parallel beams 192 inches apart
6	1	Four points cornered at a 132-in. by 108-in. rectangle

(U) Three different thrust structure designs, all using a cruciform engine installation, were evaluated for Cases 1 and 3. All of the thrust structure configurations studied were deflection limited. The maximum deflection in the structure occurs at the mount point of the engines and is 0.625 in. for Cases 1 and 3. The stiffness of each structural member was optimized with respect to the total structure weight, and the basic cone angle is optimum for all locations and each configuration.

(U) The selected mounting scheme for Cases 1 and 3, shown in figure 121, is a truncated cone within a truncated cone with crossbeam stabilizing members. In this design, the external thrust structure shape is a truncated cone, which is formed by aluminum trusses reinforced by honeycomb. The four outboard engines are mounted at the apex of the aluminum trusses. Part of the radial load is removed by aluminum crossbeams at the engine end of the thrust structure. In addition, these crossbeams provide cone stabilization. The crossbeams tie the outboard engines to the center engine, which is mounted at the intersection of the crossbeams. Additional support for the center engine is provided by the inner cone, which transmits the engine load to the mount flange. Ring stiffeners were required at the mount flange to restrain radial deflection and permit only axial loads to be transmitted to the vehicle mount structure. This structure was the lightest weight of the three configurations studied. The other two configurations were (1) a truncated cone within a truncated cone arrangement, and (2) a cone crossbeam arrangement.

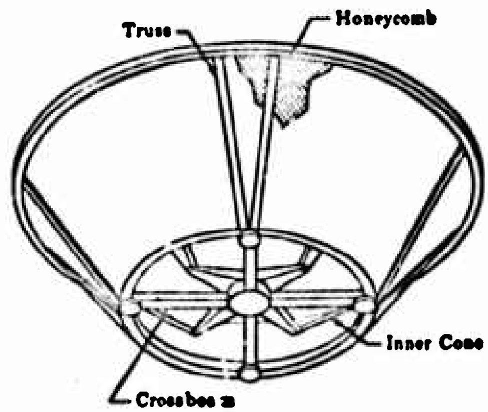


Figure 121. Thrust Structure Sketch for Cases 1 and 3 FD 20751A

(U) The cone within a cone design was similar to the selected configuration but did not have the lower support beam. The thrust of the four outboard engines was transmitted to the vehicle by the outer cone and the inboard engine thrust was transmitted through the inner cone. Each cone was sized independently for a spring constant of 2×10^6 lb/in. The cone crossbeam arrangement was also similar to the selected design but did not have the inner cone. The thrust of all five engines was transmitted to the vehicle through the single cone. Both of these configurations were appreciably heavier than the selected design.

(U) The parametric weight data, based on the designs for Cases 1 and 3, are presented in figures 122 and 123, respectively. Mount height is defined as the distance from the vehicle attachment to the engine mount plane. As the mount ring diameter is increased less bending loads are imposed, allowing a lighter structure.

(U) The thrust structure for Case 2, shown in figure 124, consists of an aluminum-tapered disk-beam bending member, truncated cone, stiffening ring, and the necessary mounting hardware. The disk-beam transmits the thrust loads radially from the single engine to the cone base. The disk-beam is composed of a tensile-loaded top plate, compressive-loaded bottom plate, and shear carrying webs. This construction technique transmits the load uniformly into the cone while allowing beam stability with thin members. The cone carries the thrust load from the outer periphery of the disk-beam member of the vehicle thrust ring. A skin-stringer construction was selected for the cone to provide a lightweight compression member. A stiffening ring and mounting flange are required at the vehicle end of the cone to prevent radial deflection and to transmit a uniformly distributed axial load to the vehicle thrust ring. Boron filament winding is used for the restraining ring material because of its high modulus-to-density ratio. The cone angle was optimized for each engine location. The parametric weight data based on this design are shown in figure 125 as a function of engine mount height. The dominating factor in sizing the structure was the deflection restraint set by the 2.0×10^6 lb/in. spring constant. The maximum deflection is 0.125 in. and occurs at the engine mount point of the disk-beam.

(U) Mounting hardware includes (1) a section to distribute the thrust loading at the disk center, (2) increased structure thickness at propellant line and gimbal actuator cutouts, and (3) stiffening webs for tieing the disk-beam into the cone.

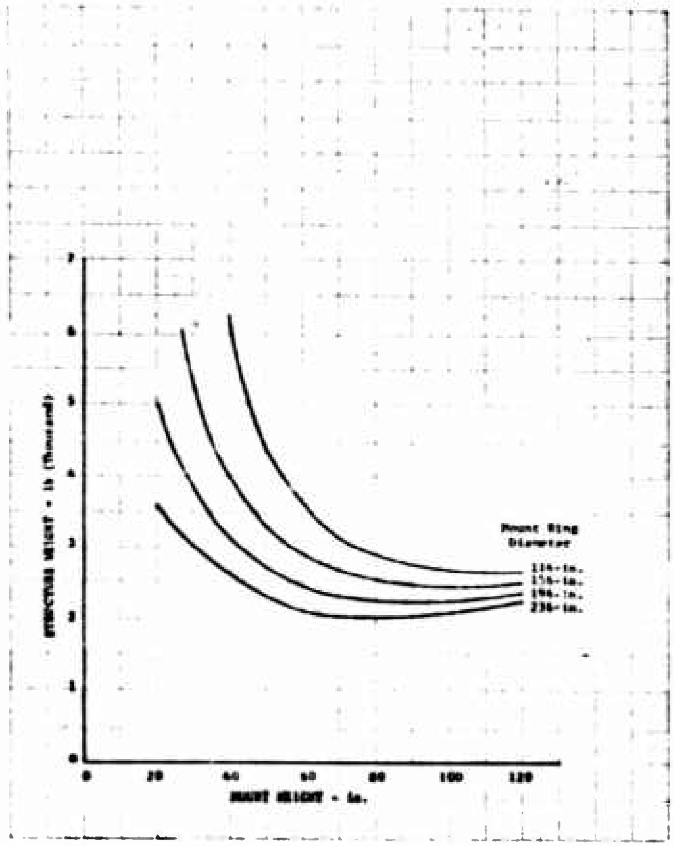


Figure 122. Thrust Structure Weight vs Mount Height for Case 1, 250K Module DF 55633

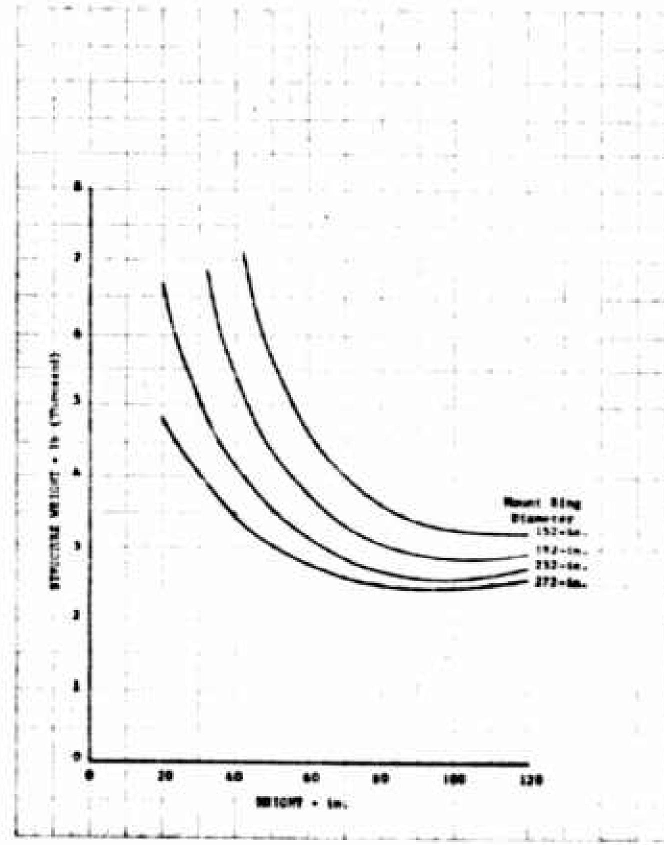


Figure 123. Thrust Structure Weight vs Mount Height for Case 3, 250K Module DF 55634

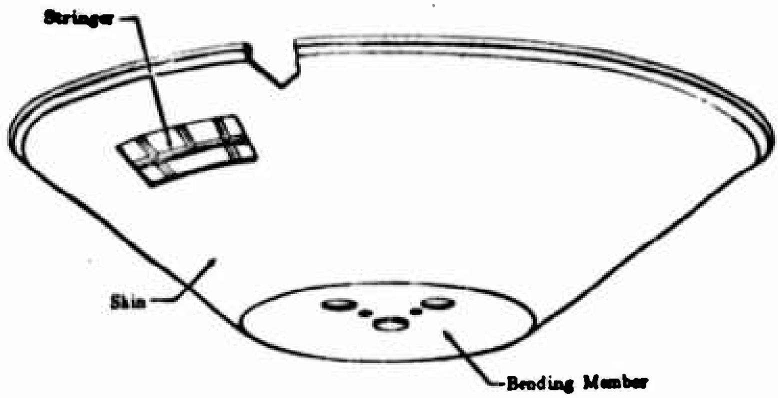


Figure 124. Thrust Structure Sketch for Case 2 FD 20080

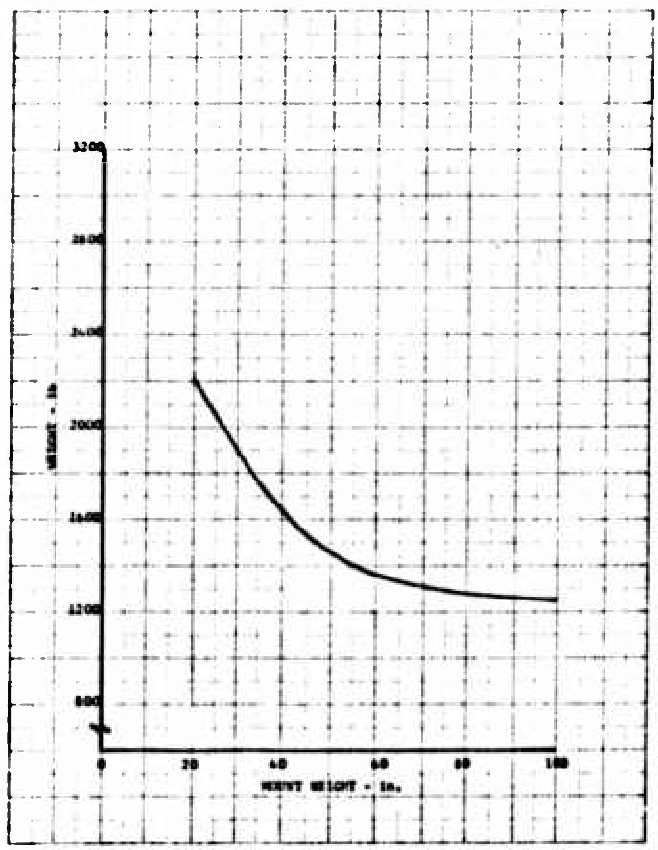


Figure 125. Thrust Structure Weight vs Mount Height for Case 2, 250K Module DF 54881

(U) The thrust structure for Case 4, shown in figure 126, is a cone-beam combination. The thrust structure is the shape of a truncated cone, which is formed by aluminum trusses reinforced by honeycomb. The eight engines are mounted at the apex of the aluminum trusses. The engines are tied together by beams that extend radially from the vehicle centerline to each engine mount point. For the few axial positions where the tank prevents this arrangement, the beams are attached to each engine mount position to form an octagon. This crossbeam arrangement was approximately 6% lighter than a configuration using only the engine mount ring as the cone base. The structure for Case 4 was either stress limited or deflection limited depending on the engine mounting location (axial and radial distance relative to the main vehicle mount ring). For the optimum location determined from the installation study, the maximum deflection is less than the allowable of 1.0 in. The parametric weight data based on this design are shown in figure 127 as a function of engine mount height.

(U) The dominating factor in sizing the thrust structure for Cases 5 and 6 was the deflection restraint set by the 2.0 x 10^6 lb/in. spring constant. The maximum deflection occurring in the structure is 0.125 in. at the engine mounting point for both Cases 5 and 6. Case 5 has two engines, however, each engine has an independent thrust structure. The structures consist of aluminum bending, compressive and tension members, shear webs, and mounting hardware. Weights have been optimized at each height to obtain the most advantageous combination of bending and compressive members.

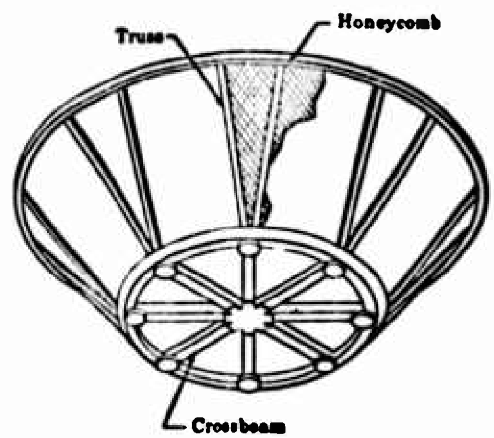


Figure 126. Thrust Structure Sketch for Case 4 FD 20752A

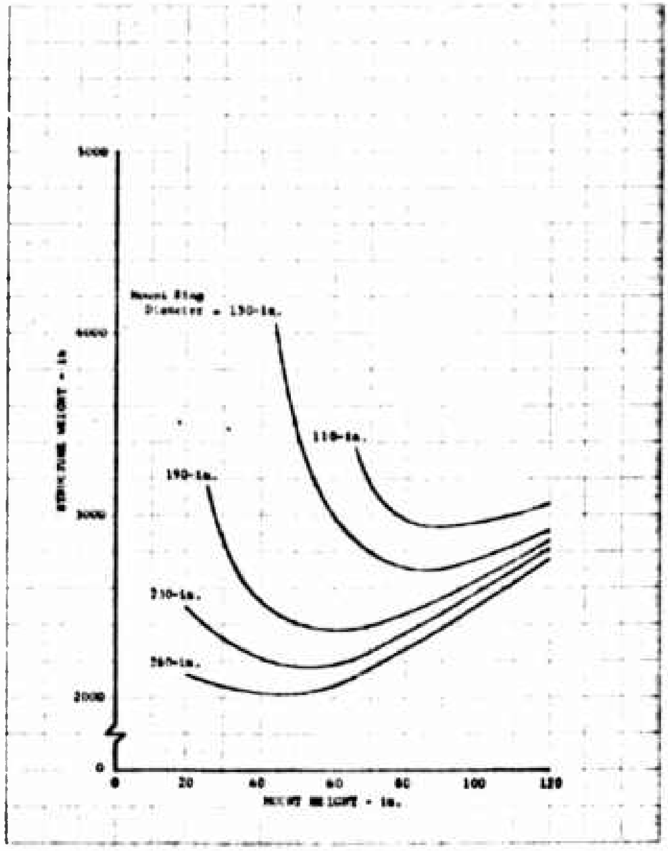


Figure 127. Thrust Structure Weight vs Mount Height for Case 4, 250K Module DF 55635

(U) The selected thrust structure for Case 5 is shown in figure 128 and for Case 6 in figure 129. The structures are similar and consist of aluminum tapered beam bending members, columns, honeycomb sheets, tension members, and mounting hardware. The tapered crossbeams transmit the thrust load to the columns at the beam ends. The columns transfer the thrust load to the vehicle mount points. Tension members are used to absorb side loads and to transmit the engine thrust to the vehicle in the axial direction only. Honeycomb webs are used between the columns to provide shear stiffness during engine gimbaling. The parametric thrust structure weights based on this design are shown in figures 130 and 131 for Cases 5 and 6, respectively.

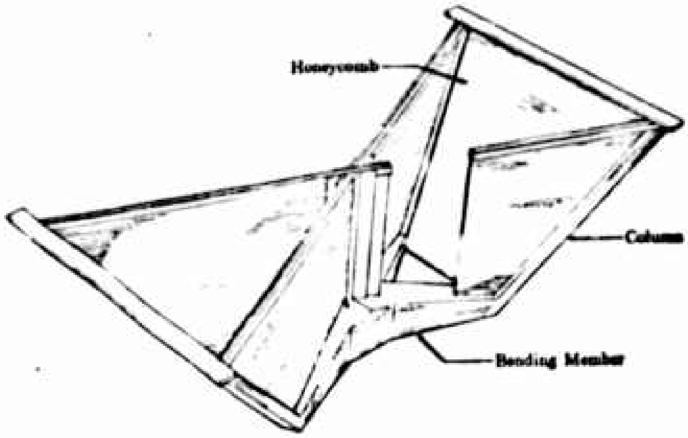


Figure 128. Thrust Structure Sketch for Case 5 FD 20072

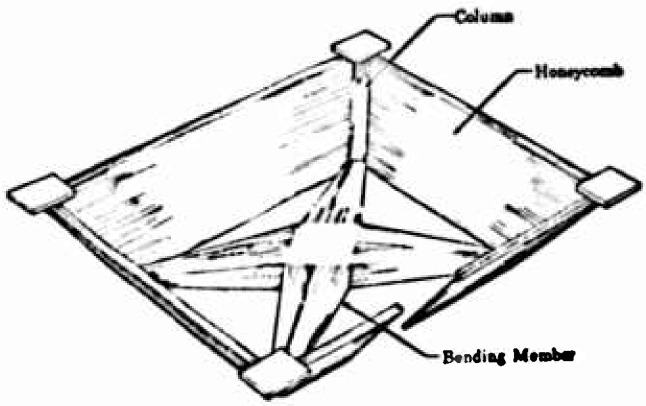


Figure 129. Thrust Structure Sketch for Case 6 FD 20073

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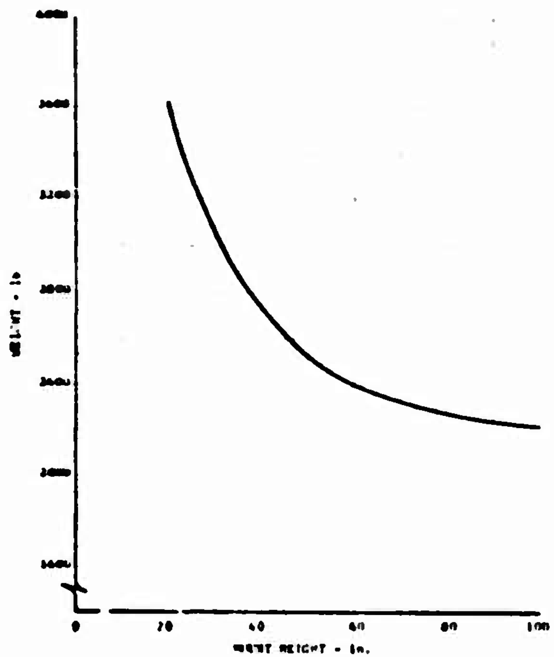

Figure 130. Thrust Structure Weight vs Mount Height for Case 5, 250K Module DF 54880

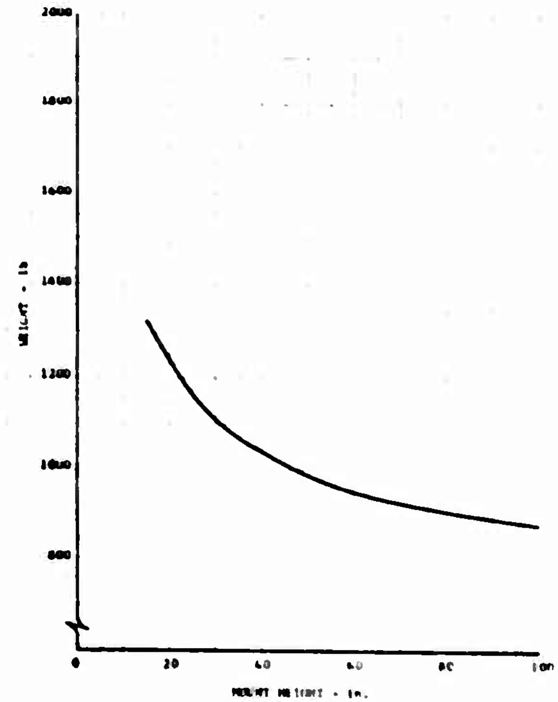

Figure 131. Thrust Structure Weight vs Mount Height for Case 6, 250K Module DF 54879

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2. 350K Module

(U) The number of engines and the supplied vehicle mount for each of the six cases using the 350K engine module is presented in the following list.

Case	Number of Engines	Supplied Vehicle Mount
1	4	288-in. diameter ring
2	1	328-in. diameter ring
3	4	324-in. diameter ring
4	6	276-in. diameter ring
5	1	Parallel beams 168-inches apart
6	1	Four points cornered at a 120-in. by 148-in. rectangle.

(U) The lower stage cases (Cases 1, 3, and 4) using the 350K engine module differ from the 250K cases in vehicle mount diameter, stage thrust, allowable deflection, and number of engines. Because 350K cases do not use the cruciform engine arrangement, the selected thrust structure concept is a truncated cone with crossbeams similar to the 250K Case 4 previously described. Where structures were sized by the spring constant (2.0×10^6 lb/in.) limit, the stiffness of each member was optimized with respect to the total structure weight. The specified spring constant sized the structure for Cases 1 and 3 and resulted in a maximum deflection of 0.700 in. at the engine mounting point. Case 4 was both stress and deflection limited depending on the mounting location. For the selected location the maximum deflection is less than the allowable of 1.05 in. The parametric thrust structure weights calculated for the lower stage cases are shown in figures 132, 133, and 134 for Cases 1, 3, and 4, respectively.

(U) The 350K engine thrust structure designs are similar to the 250K designs for the upper stage configurations. The vehicle mount dimensions, stage thrust, and allowable deflection are the major differences between the two studies. Both Cases 2 and 6 use the respective thrust structure design selected for the 250K module analysis previously discussed. The parametric thrust structure weights for Cases 2 and 6 are shown in figures 135 and 136. These structure designs are controlled by the 2.0×10^6 lb/in. spring constant and result in a maximum deflection of 0.175 in. at the engine mount point.

(U) Because Case 5 is a single 350K module installation, the 250K Case 6 design was modified to transmit the thrust load to the parallel beams of Case 5. The parametric thrust structure weights are shown in figure 137 for Case 5. The structure is also deflection limited and results in a maximum deflection of 0.175 in. at the engine mounting point.

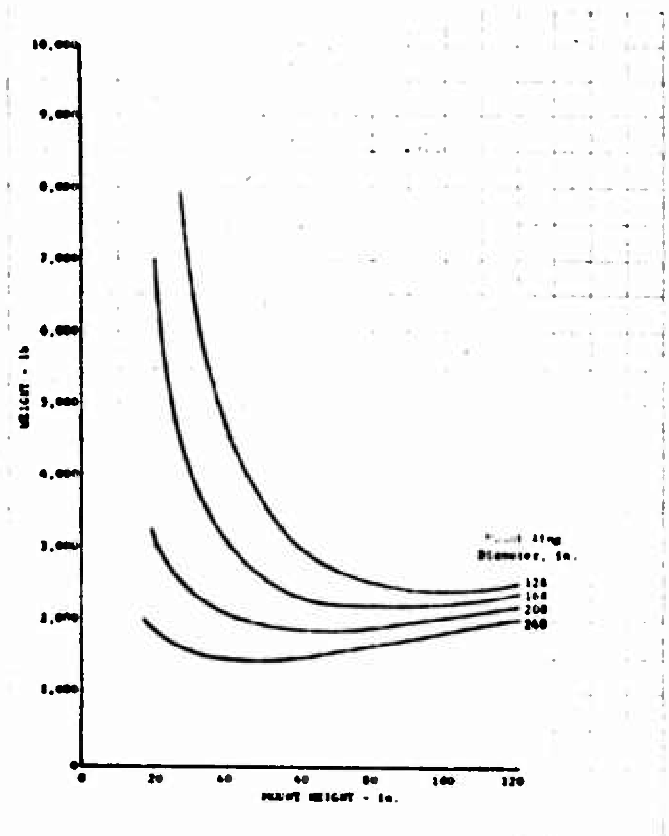

Figure 132. Thrust Structure Weight vs Mount Height for Case 1, 350K Module — DF 57128

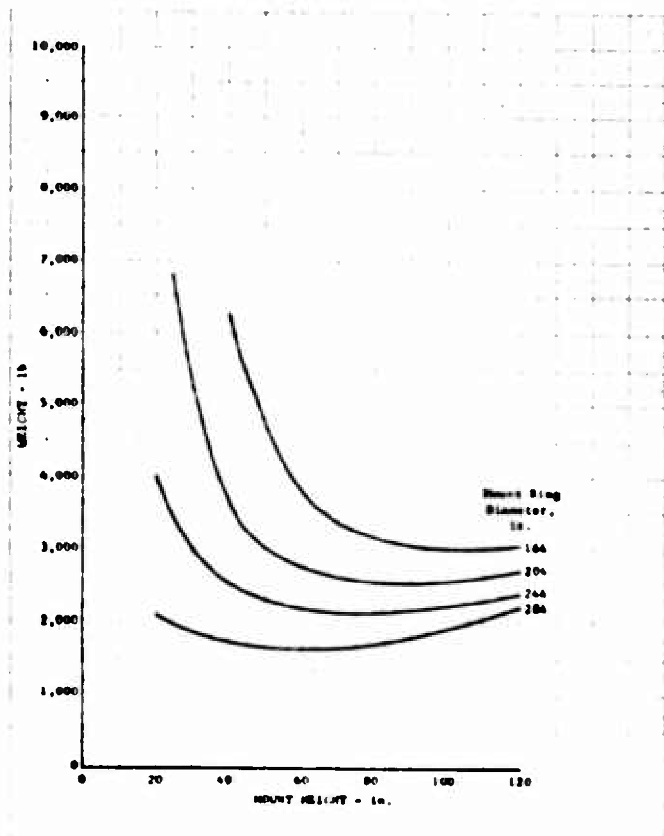

Figure 133. Thrust Structure Weight vs Mount Height for Case 3, 350K Module — DF 57129

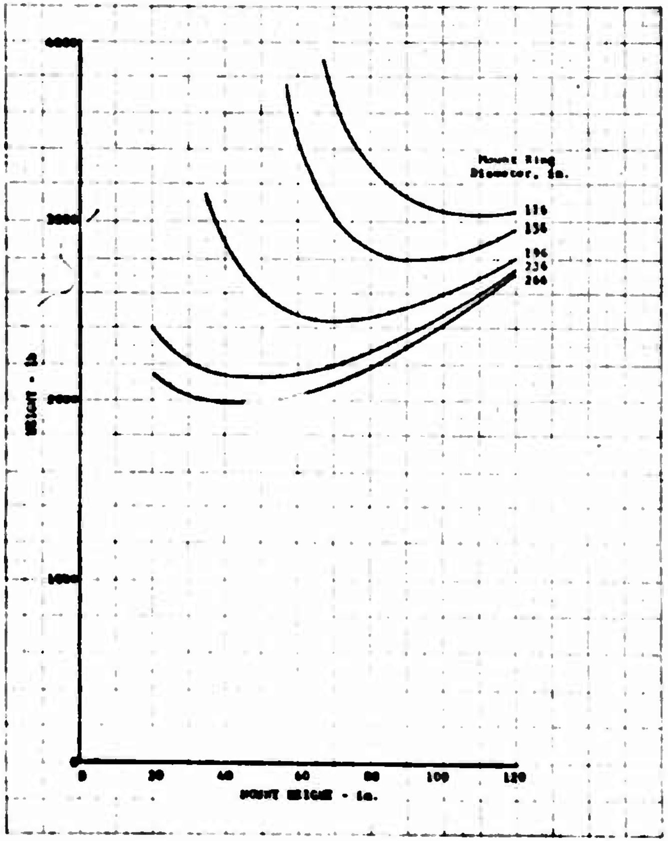

Figure 134. Thrust Structure Weight vs Mount Height for Case 4, 350K Module — DF 57130

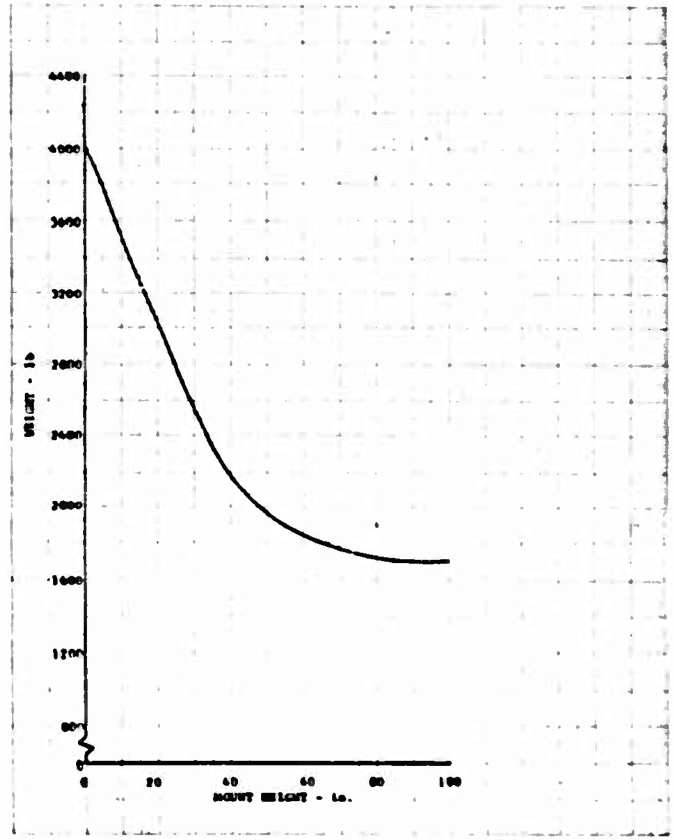

Figure 135. Thrust Structure Weight vs Mount Height for Case 2, 350K Module — DF 57125

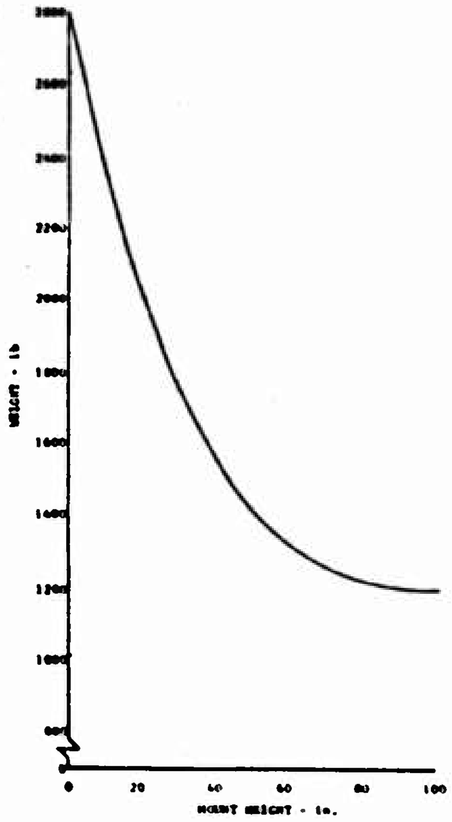

Figure 136. Thrust Structure Weight vs Mount Height for Case 6, 350K Module DF 57126

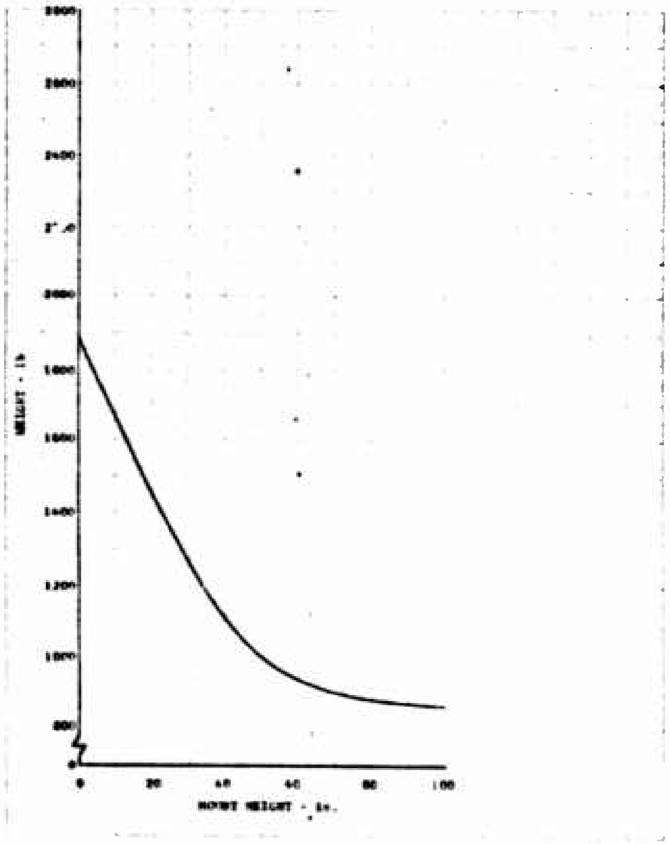

Figure 137. Thrust Structure Weight vs Mount Height for Case 5, 350K Module DF 57127

(U) For the analysis of the required thrust structure the following material properties were used.

Aluminum:

Ultimate strength	68,000 lb/psi
Safety factor	1.5
Modulus of elasticity	10×10^6 psi
Density	0.101 lb/in.

Boran filaments:

Ultimate strength	150,000 lb/psi
Safety factor	1.5
Modulus of elasticity	39.0×10^6 psi
Density	0.071 lb/in.^{3}

B. PROPELLANT FEED LINES

(U) The feed lines provide the propellant passage from the specified tank discharge location of each application case to the engine fuel and oxidizer inducer inlets.

(U) The components considered in the propellant feed line weights are (1) the propellant line, (2) vacuum jacket, (3) an omnidirectional bellow (provided at the inducer inlet to accommodate engine gimbaling), (4) flexible joints (for expansion and relative movement between feed line and attach structure), (5) a diffuser (transition from propellant line diameter to larger inducer inlet diameter), (6) the required attaching flanges, and (7) Teflon standoffs. These are shown in figure 138. Stiffeners were integrated into the vacuum jacket line. The stiffeners used are circumferential ridges around the vacuum jacket. The stiffeners and standoffs were used to maintain jacket integrity.

(U) A computer program was used to calculate feed line weight and pressure drop as a function of engine mount height (distance from the vehicle thrust mount to the engine mount plane) and the outboard engine gimbal point (lower stage cases) for various line sizes. The program determines the line routing and length for the constraints of each installation case. The pressure drop calculation includes the effect of friction, line dimensions, elbows, bellows, and diffuser. According to the specified ground rules, only the steady-state pressure loss was considered. The line weights were determined for stainless steel vacuum jacketed lines and components.

1. 250K Module

(U) The line sizes were selected for the 250K cases on the basis of installation and pressure drop considerations. Table XX summarizes the propellant feed line sizes selected for the installation optimizations.

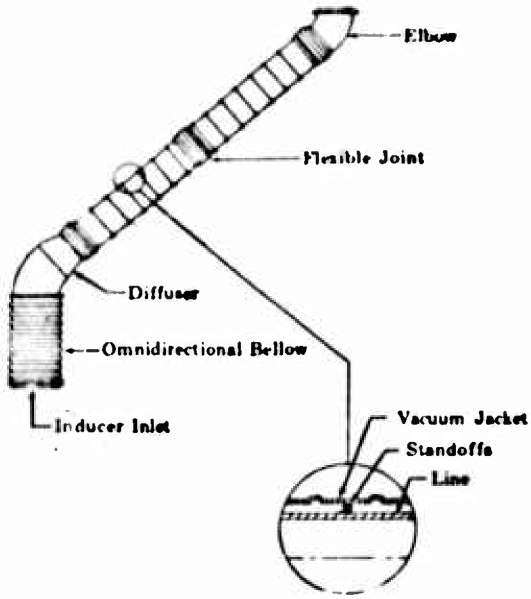

Figure 138. Typical Sketch of Feed Line FD 20095A

(U) Table XX. 250K Module Propellant Feed Line Sizes

Case	Line Size (ID), in. Oxygen	Hydrogen
1	8.5	8.0
2	8.0	8.0
3	8.5	8.0
4	8.5	8.0
5	8.5	8.0
6	8.0	8.0

(C) The larger oxygen line sizes for Case 5 was required because of increased line bends and increased pressure loss resulting from locating the engine between the propellant tanks. Eight-inch oxygen lines for Cases 2 and 6 resulted from the low pressure drop associated with the relatively straight and short line allowed by the installations. The oxygen line size for Cases 1 and 3 was dominated by the pressure loss of the center engine. Conversely, the hydrogen line size was based on the outboard engine requirements. For the above selected line sizes, the feed line pressure drop between the vehicle interface and the engine inlet does not exceed the specified steady-state pressure drop of 3.0 psi for the hydrogen lines and 2.5 psi for the oxygen lines. Maximum pressure was determined using flow rates at a mixture ratio of 5.0 for the hydrogen and 7.0 for the oxygen for all locations.

(U) The feed line weights were calculated by using the S-II stage as the base point (0.032-inch thickness at a 10-inch line diameter). This base point was used to include the effects of vibration and installation loading into the line weights. The resulting feed line wall thickness was 0.026 inch for the 8.0-inch lines and 0.028 inch for the 8.5-inch lines. The vacuum jacket thickness was held constant at 0.016 inch. The 0.016-inch vacuum jacket thickness was assumed to be the minimum gage for ease of handling and installation. The stiffener spacing was varied with line size to withstand the collapse loads.

(U) Feed line weights are shown in figures 139 and 140 for the upper stage vehicle (Cases 2, 5, and 6) as a function of engine mount height. In Case 5, the between tank mounting results in decreased line length as mount height is increased. Figures 141 through 143 indicate the weight for the lower stages as a function of engine mount height and outboard engine gimbal point radius. The parametric range shown for each case includes the limitations imposed by (1) propellant tank, (2) skirt clearance requirements, and (3) engine envelope restrictions. Clearance for the engine was determined by considering a maximum gimbal position (7 degree gimbal angle). For the multiple engine installation, envelopes were computed separately by considering a ± 7 degree axis for each engine individually.

2. 350K Module

(U) The propellant feed line parametric weight analysis for the 350K cases was based on requirements similar to those discussed for the 250K cases. The computer program used to determine the weight and pressure drop as a function of engine mount height and engine gimbal radius was modified for the 350K module application cases.

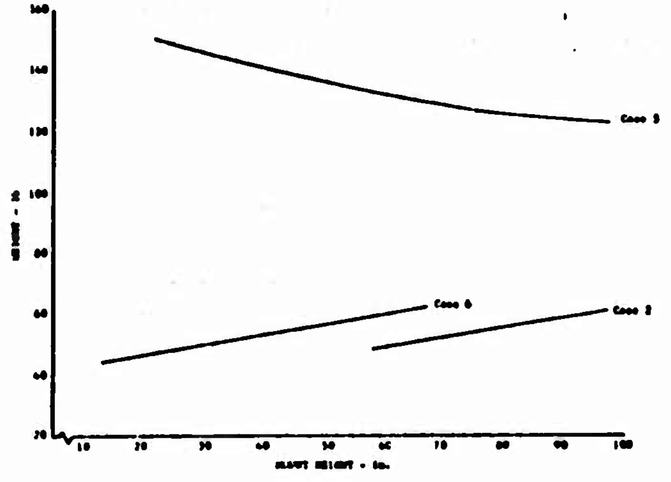

Figure 139. Hydrogen Feed Line Weights, 250K Module Upper Stages DF 54874

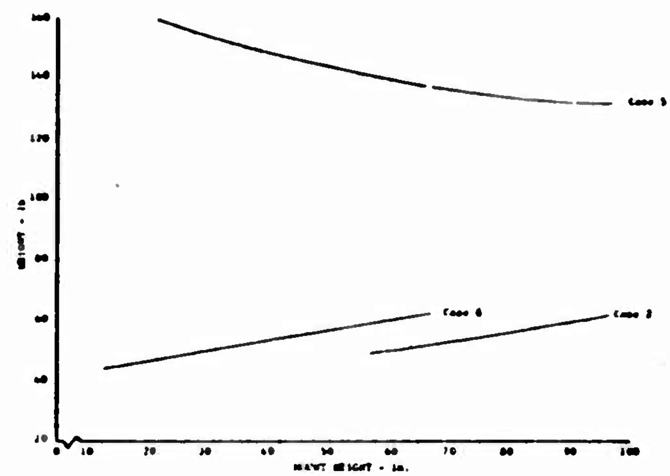

Figure 140. Oxygen Feed Line Weights, 250K Module Upper Stages DF 54875

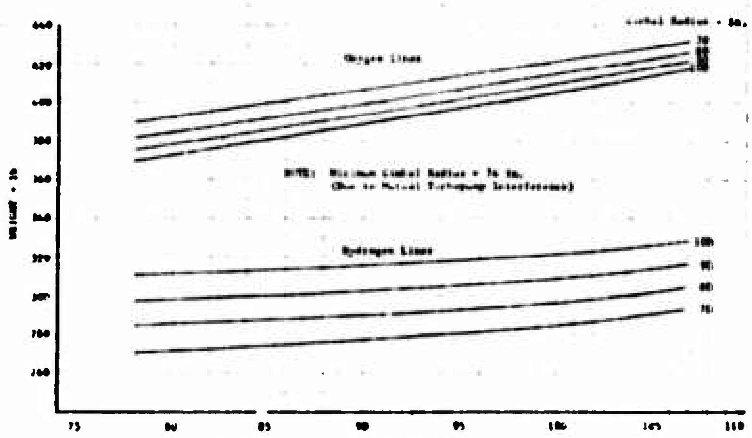

Figure 141. Case 1 Feed Line Weights, 250K Module DF 54876

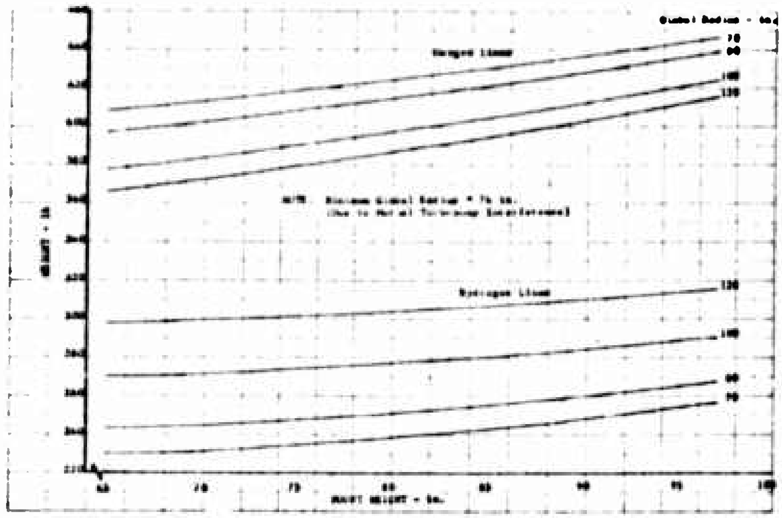

Figure 142. Case 3 Feed Line Weights, 250K Module DF 54877

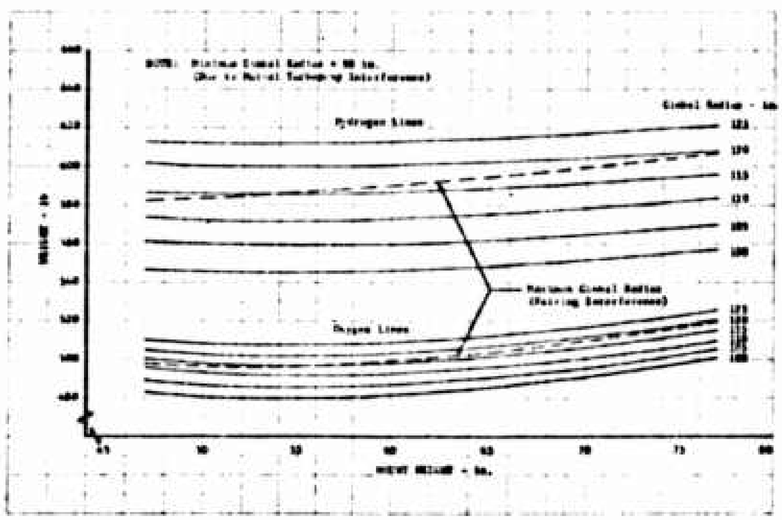

Figure 143. Case 4 Feed Line Weights, 250K Module DF 54878

(U) Feed line diameters were selected for each case on the basis of installation and pressure loss consideration. Table XXI summarizes the propellant line selected sizes.

(U) Table XXI. 350K Module Propellant Feed Line Sizes

Case	Line Size (ID), in. Fuel	Oxygen
1	9.5	10.0
2	9.5	9.5
3	9.5	10.0
4	9.5	10.5
5	9.5	9.5
6	9.5	9.5

(C) For the selected line size, the maximum pressure loss between the propellant tank outlet and the engine inlet does not exceed the specified steady-state pressure loss of 3.0 psi for the hydrogen line and 2.5 psi for the oxygen lines. Pressure loss was calculated by using a flow rate consistent with mixture ratios of 5.0 for the hydrogen and 7.0 for the oxygen. The primary loss was determined for all of the engine mounting positions and the selected line size was based on the worst condition.

(U) Propellant feed line weights are shown in figures 144 and 145 for the upper stage vehicles (Cases 2, 5, and 6) as a function of engine mount height. Case 5 has two hydrogen tanks and, therefore, two hydrogen outlets that are located at considerable distance (approximately 14 ft) from the engine mounting point. This results in the relatively heavy hydrogen feed line weights shown in figure 144 for the 350K module installation. Figures 146 through 151 show the feed line weights for the lower stage cases (Cases 1, 3, and 4) as a function of engine mount height and engine gimbal point radius. The parametric range shown for each case includes the limitations imposed by (1) the propellant tank, (2) skirt clearance requirements, and (3) engine envelope requirements. The weight includes the components shown in figure 148.

(U) Using the same ground rules as discussed for the 250K module, the wall thickness was 0.031 in. for the 9.5-in. line, 0.032 in. for the 10.0 in. line, and 0.033 in. for the 10.5-in. line.

C. HEAT SHIELD

(U) A heat shield is required on the lower stages (Cases 1, 3, and 4) to protect the vehicle from recirculated exhaust gases. The heat shield must withstand base pressure loading of ± 1 psi and limit the engine compartment temperature to less than 960°R.

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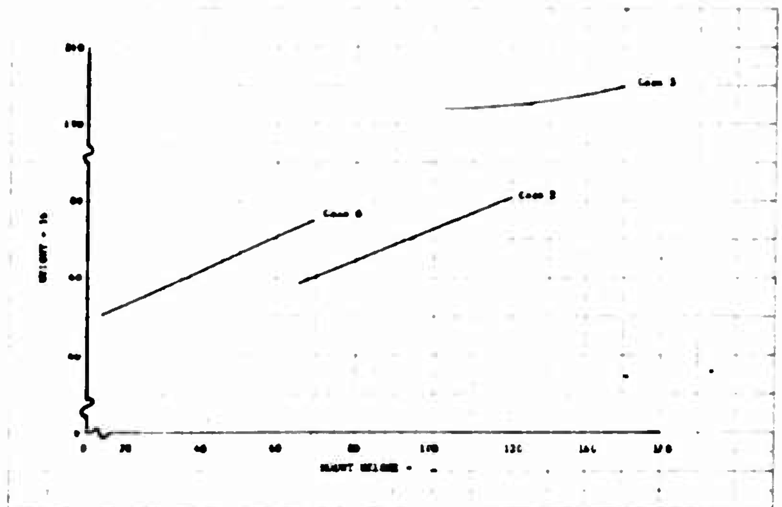

Figure 144. Hydrogen Feed Line Weights, 350K Module Upper Stages DF 57131

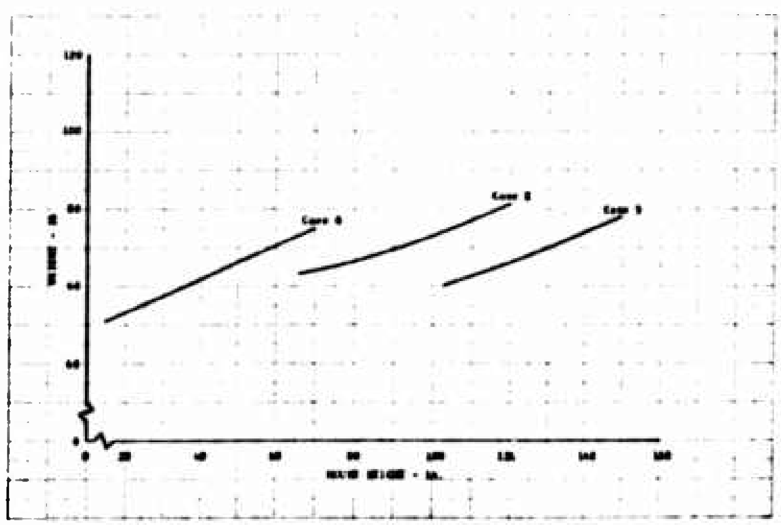

Figure 145. Oxygen Feed Line Weights, 350K Module Upper Stages DF 57132

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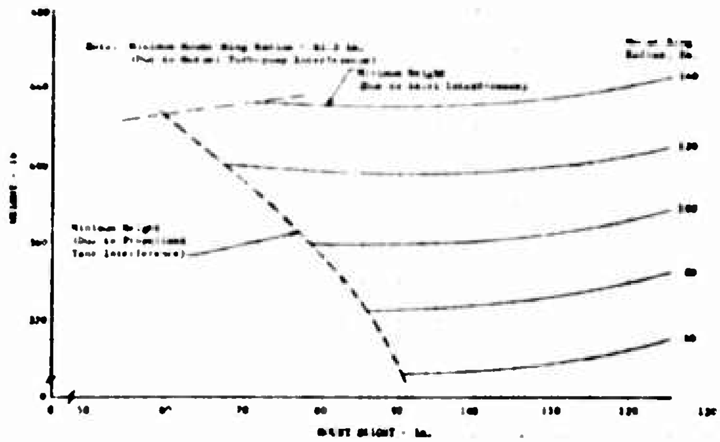


Figure 146. Case 1 Hydrogen Feed Line Weights, 350K Module DF 57133

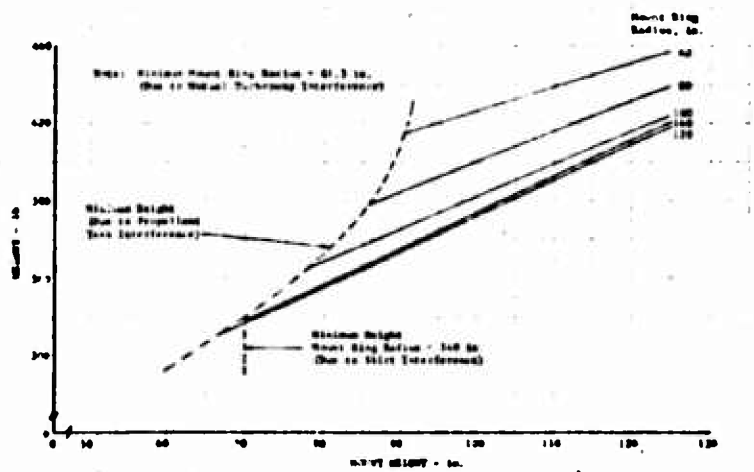


Figure 147. Case 1 Oxygen Feed Line Weight, 350K Module DF 57134

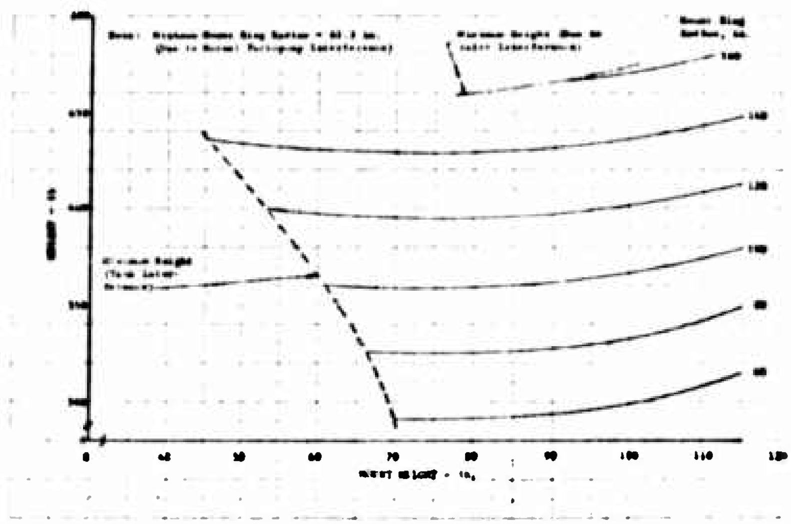


Figure 148. Case 3 Hydrogen Feed Line Weights, 350K Module　　DF 57135

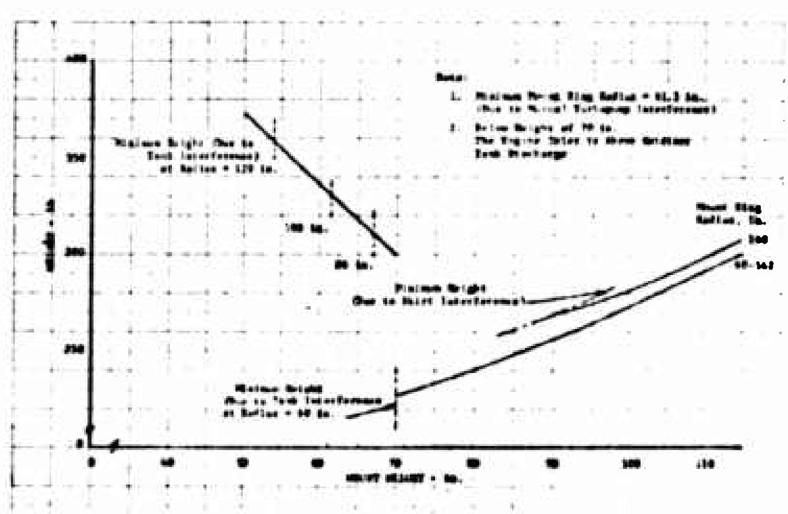


Figure 149. Case 3 Oxygen Feed Line Weights, 350K Module　　DF 57136

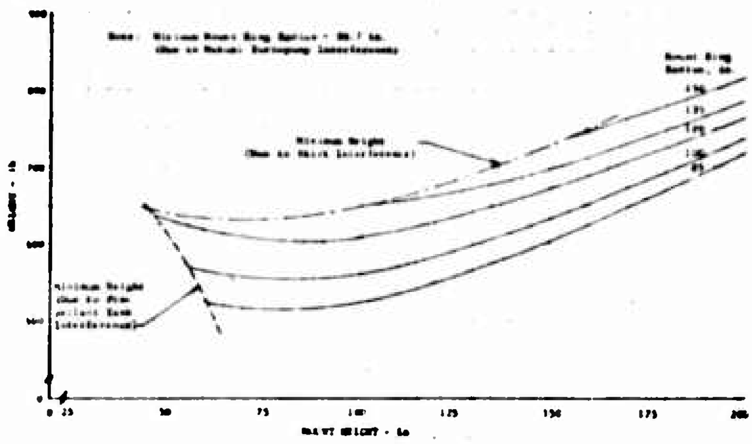


Figure 150. Case 4 Hydrogen Feed Line Weights, 350K Module DF 57137

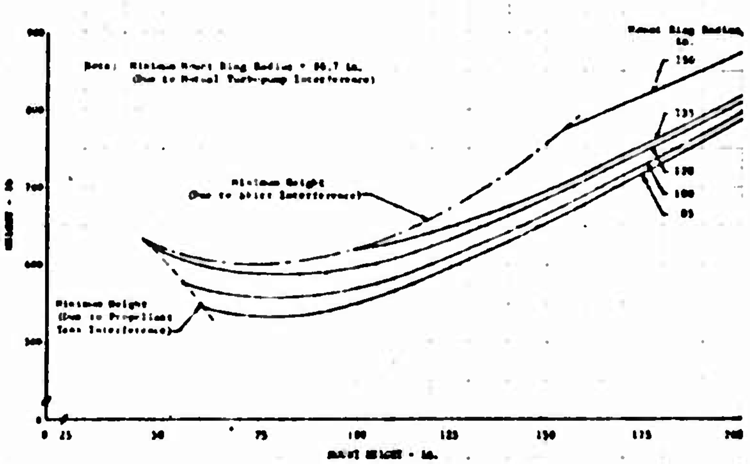


Figure 151. Case 4 Oxygen Feed Line Weights, 350K Module DF 57138

(U) The heat shield was mounted at the engine and completely enclosed the vehicle base by extending to the vehicle fairing. Base pressure loading of ± 1 psi was carried by aluminum honeycomb, supported by aluminum beams tied by a structural member into the main thrust structure. Allowable stress in the honeycomb and structure supports was based on an operating temperature of 660°R. A thermal protection barrier of approximately 1.5 inches of silica fiber insulation protects the load carrying honeycomb from the thermal environment. The insulation was sized, based on the results of a detailed transient heat transfer analysis of the complete heat shield structure, to prevent the aluminum honeycomb from exceeding its design temperature during the vehicle boost.

(U) Base heating is caused by both convective and radiative heat transfer modes and is a function of altitude. Because of the complex configuration and environment around the base of a multiengine vehicle, a purely theoretical analysis of base heating is not satisfactory. Therefore, the heating environment to be used in the design of a base heat shield is generally obtained through model tests, and empirical relations developed through experience. The heating environment used in this analysis (figure 152) was obtained from flight and model test experience for the S-IV, S-II, and S-IB stages. The base heating rate varied from 31 Btu/ft^2 sec at sea level to 6 Btu/ft^2 sec at altitude. The exhaust gas recovery temperature was 3250°R and constant with altitude.

(U) The transient calculations were performed by the numerical finite difference method for the thermal model shown in figure 153. Because the heat shield diameter is quite large compared to its thickness, the heat shield was handled as an infinite plate (i.e., temperature is only a function of thickness).

(U) The initial and convective boundary conditions were as follows:

1. Hot-side heating environment as function of time
2. Surface radiation as function of surface temperature
3. Cold side - free convection h = 2.5 Btu/ft^2 hr °R
4. Heat shield temperature at the start of stage operation uniform at 540°R.

(U) The insulation used is composed of 99% pure silica fiber, has a density of approximately 4.5 lb/ft^3 and will withstand steady-state temperatures of 3200°R and up to 3450°R for transients. The thermal conductivity for this material is a function of temperature and varies in values from 0.5 Btu in./ft^2 lb °R at 1000°R to 1.5 Btu in./ft^2 lb °R at 2500 °R.

(U) A weight breakdown of the main heat shield components is provided in table XXII for the 250K cases. In all cases, the heat shield extends to the vehicle fairing. The total heat shield weight as a function of radius is shown in figure 154. Figure 154 applies to both 250K and 350K cases

because an allowance for the engine circular cut-out was not subtracted from the weight. Due to the longer burn time of Case 3, increased insulation thickness is required and therefore results in a slight increase in weight as shown in figure 154.

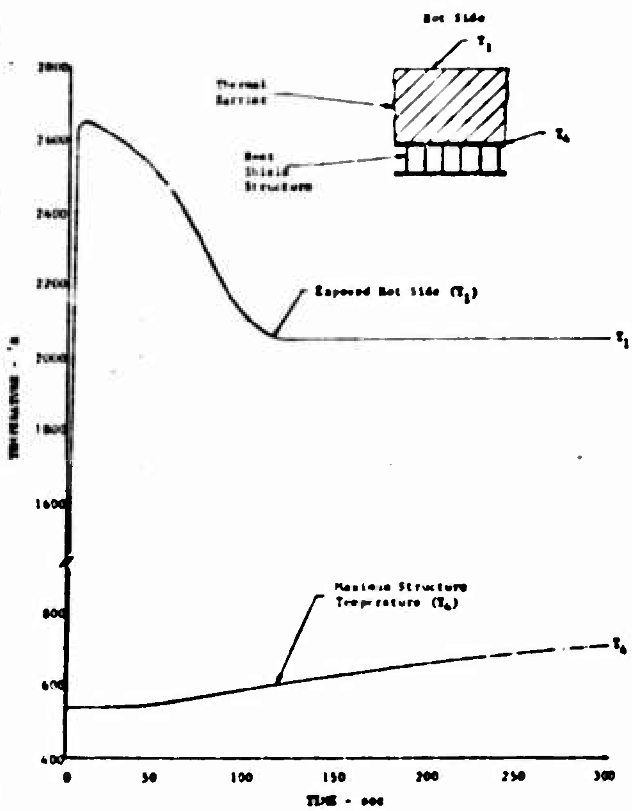
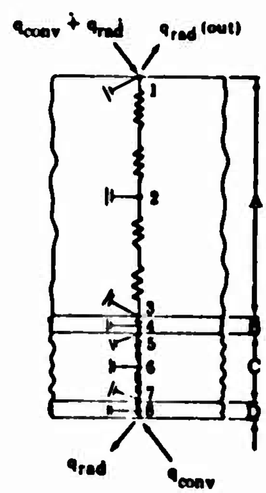


Figure 152. Estimated Heat Shield Temperature History DF 57264

(U) Table XXII. Heat Shield Weight*

Case	Total, lb	Honeycomb, lb	Mounting, lb	Insulation, lb
1	760	325	165	270
3	1012	415	185	412
4	722	312	162	248

*Heat shield radius equal to vehicle radius.



Location	Dimension	Material
A	As Required	J-M Dyna - Quartz
B	0.020 in.	Aluminum
C	0.250 in.	Aluminum Honeycomb
D	0.020 in.	Aluminum

Figure 153. Thermal Model for Transient Calculations FD 21494

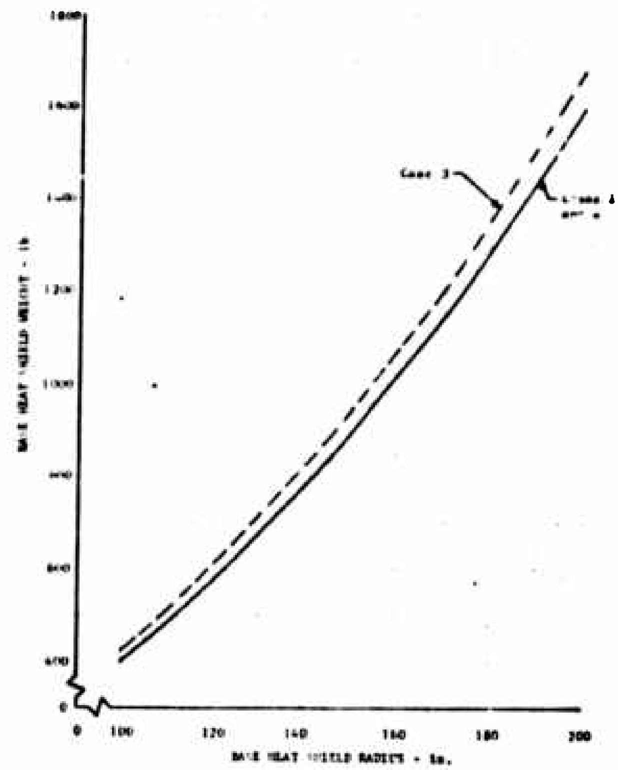


Figure 154. Heat Shield Weight for Cases 1, 3, and 4 DF 57139

D. PRESSURIZATION SYSTEM

(U) The engine associated weight of a propellant tank pressurization system is required as part of the installation weight term (W_e) of the Performance Index Calculation (W_x). The pressurization systems utilized in the Saturn S-IV, S-IVB, S-II, and MS-II vehicles were reviewed. This review indicated the desirability of using gaseous hydrogen and gaseous oxygen as pressurants.

(U) The pressurization selected for all of the application cases was engine supplied gaseous hydrogen and gaseous oxygen obtained from an engine supplied heat exchanger. In accordance with the supplied Applications Package (Appendix I) the weight estimates used in the W_x equation are engine associated hardware only. Plumbing was included to deliver the pressurants to a vehicle interface defined as the main vehicle structure ring.

1. 250K Module

(U) The required pressurant flow rates from each engine for the propellant tank pressurization system selected are shown in figure 155 as a function of the gas supply temperature. These values were determined for collapse factors of three for hydrogen and two for oxygen. The required gaseous hydrogen was considered to be trapped from the transpiration supply heat exchanger. The temperature available at this point resulted in a gaseous hydrogen flow rate of 0.85 lb/sec per engine. The required gaseous oxygen is supplied by the engine through an engine mounted heat exchanger. The selected heat exchanger configuration resulted in a gaseous oxygen flow rate requirement for tank pressurization of 2.5 lb/sec per engine.

(U) Table XXIII summarizes the pressurization system weights for Cases 1 through 6.

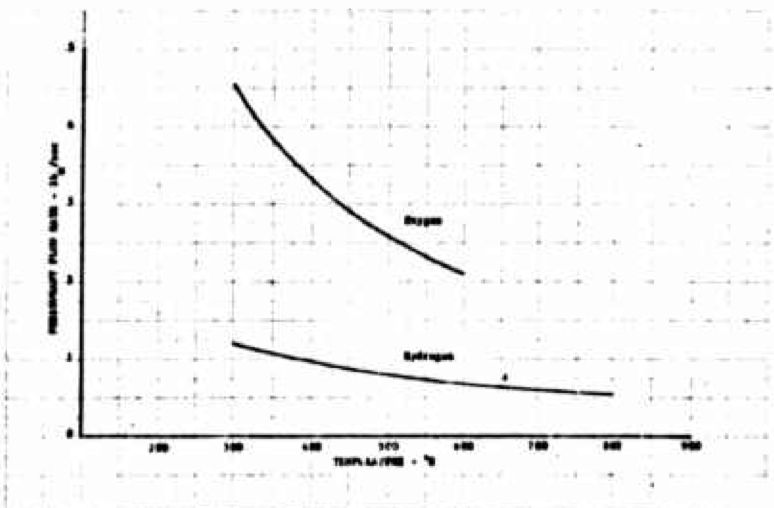


Figure 155. Pressurant Flow Rate vs Temperature, 250K Module DF 55702

(U) Table XXIII. Pressurization System Weight, 250K Module

Case	Total Pressurization System Weight, lb
1	154
2	24
3	151
4	256
5	48
6	24

(U) The pressurization system weights, shown in table XXIII, include engine associated hardware and component systems only. The weight estimates for the lower stages include manifolds and the necessary plumbing to accommodate the multiple engine applications (Cases 1, 3, and 4).

(U) The engine associated hydrogen system is composed of (1) a throttling and shutoff valve, and (2) the necessary plumbing to transport the gaseous hydrogen from the engine extraction point to the vehicle interface, which was considered to be the thrust structure mount point. Line and valve sizes were based on the selected hydrogen flow rate of 0.85 lb/sec. An acceptable gaseous hydrogen flow velocity was used to determine the single engine line size (1.250-inch ID).

(U) The engine associated oxygen system is composed of (1) a shutoff valve, (2) throttling and shutoff valve, and (3) an engine mounted heat exchanger and required plumbing. Gaseous oxygen lines and valves were sized for a flow rate of 2.5 lb/sec.

(U) For the lower stage applications (Cases 1, 3, and 4), pressurant for each propellant tank is delivered by a main pressurant line, running from a manifold located near the gimbal mount plane to the vehicle-engine interface. For the lower stages the pressurization system weights shown in table XXIII include manifolds and main pressurant lines as well as those components previously outlined.

(U) A heat transfer analysis was performed to estimate the required heat exchanger configuration. The heat exchanger was considered to be located in the preburner transition case just upstream of the main injector. The pressurization flow was taken from the engine at the oxidizer injector inlet where the pressure and temperature are approximately 3500 psia and 200°R. Stress limits of the tubular heat exchanger were established by dry operation. The pressurization flow rate studied varied from 2 to 6 lb/sec. The heat exchanger area did not vary significantly with pressurant flow rate, the heat exchanger weight was relatively insensitive to this parameter. Because the pressurization flow is obtained from the injector inlet at high pressure, a throttle valve must be placed in the system to reduce the pressure of the oxygen before it can be used for pressurization. Considering the effect of the pressure level on thermodynamic and transport properties of the oxygen, it was determined that the optimum location for the throttle valve was downstream of the heat exchanger. For a given bulk pressurant velocity, the net heat input to the oxygen for an exchanger operating at 3500 psia is twice that at 400 psia. The strong influence of pressure level on transport properties affects the increased heat strength required due to throttling the gaseous discharger.

2. 350K Module

(U) The required pressurant flow for the 350K module cases is 1.2 lb/sec per engine for the fuel system and 3.5 lb/sec per engine for the oxygen system. Table XXIV summarizes the pressurization system weights for Cases 1 through 6. The components are similar to those discussed for the 250K module. Line sizes were established by obtaining the same fluid velocity and wall stress as the 250K plumbing.

(U) Table XXIV. Pressurization System Weight, 350K Module

Case	Total Pressurization System Weight, lb
1	15[illegible]
2	28
3	150
4	236
5	28
6	28

E. FAILURE DETECTION SYSTEM

(U) The engine portion of the failure detection system provides the function of sensing abnormal engine operation by measuring the appropriate temperatures, pressures, and propellant flow rates. A review of the READI study[1] results has provided a basis for estimating system requirements and weights.

(U) The Weight of the failure detection system was estimated to be 55 pounds per engine. The system weights used in the performance index calculations are shown in table XXV.

(U) Table XXV. Failure Detection System Weights

Case	250K Module lb	350K Module, lb
1	275	220
2	55	55
3	275	220
4	440	330
5	110	55
6	55	55

F. THRUST VECTOR CONTROL SYSTEM

(U) Mechanical gimbal and secondary gas injection systems were investigated for the thrust vector control. The comparison of these two systems (Section V) showed the secondary injection system to be considerably heavier than a mechanical system. The system requirements are to provide a gimbal angle of ± 7 degrees, an angular acceleration of 30 rad/sec^2 and an angular velocity of 30 deg/sec. A review of current operational systems led to the selection of a hydraulic mechanical gimbal system powered by an engine driven hydraulic pump.

[1] "Rocket Engine Analyzer and Decision Instrumentation (READI) Investigation (Phase I)", Report No. CA-4251-0160, Sperry Gyroscope Co., December 1962.

(U) The actuation system consists of two servoactuators, a main hydraulic pump, an accumulator-reservoir, an auxiliary pump and motor, and the necessary fluid, plumbing, and installation hardware. The actuators provide the required gimbal force and supply position feedback to the closed-loop vehicle control. The actuator position is controlled by a servovalve. The main hydraulic pump supplies the high pressure fluid (3000 psia) required by the actuators and is driven by the accessory pad on the engine fuel boost pump. The accumulator-reservoir provides smooth system operation by preventing pressure surges and replenishes lost fluid. The auxiliary pump and motor provides power to move the engine to the null position before ignition and in the event of main system failure. The auxiliary system also keeps the fluid warm during extended engine shutdown by recirculating the fluid through the system.

(C) The engine selected for the TVC analysis has a two-position nozzle with an area ratio of 184. This size engine gives a representative TVC weight for both the lower and upper stage application cases. The engine moment of inertia about the gimbal point was determined with the nozzle in the extended position. The moment of inertia values determined for the study engines are shown in table XXVI.

(U) Table XXVI. Engine Moment of Inertia

Plane	Moment of Inertia (slug/ft^2) 250K	350K
Roll (I_{xx})	215	260
Pitch* (I_{yy})	850	1320
Yaw (I_{zz})	870	1355

*Pitch plane defined in direction of preburner.

(U) The items considered in sizing the system was the force required to, (1) accelerate the moment of inertia at 30 rad/sec^2, (2) overcome inlet bellows and gimbal friction resistance, (3) resist forces imposed by external acceleration forces and thrust misalignment. The spring rate used for the propellant line inlet bellows was 500 lb/in. The oxygen inlet bellows centerline distance from gimbal axis is 21.3 inches for the 250K module and 25.8 inches for the 350K module and the fuel bellows centerline extends 23.8 inches for the 250K module and 28.4 inches for the 350K module. The gimbal friction calculations were based on a gimbal ball diameter of 5 inches for the 250K module and 7 inches for the 350K module. The coefficient of friction for the gimbal assembly was 0.1. This value is representative of a Beryllium-Copper bearing insert against stainless steel with MLF-5 dry lubricant (PWA 566). The maximum thrust misalignment of 0.1 inch at the gimbal axis was based on RL10 experience. The maximum instantaneous loads during operation were considered to be 3 g's in the axial direction and 6 g's in the lateral direction. The torque required to offset each of the loads described above is shown in table XXVIa.

(U) Table XXVII. Thrust Vector Control Loads

Component	Torque (in. - lb) 250K	350K
Gimbal Acceleration (30 rad/sec^2)	314,000	484,000
Inlet Bellows (500 lb/in.)	46,500	65,000
Gimbal Friction	66,500	130,000
Acceleration Loads	412,000	562,500
Thrust Misalignment	25,000	35,000
Total	864,000	1,276,500

(U) The maximum required actuator forces are 33,400 lbf for the 250K module and 44,800 lbf for the 350K module. The resulting actuator piston areas were 11.2 in^2 and 14.2 in^2 for the 250K and 350K engines, respectively.

(U) The thrust vector control weights breakdown is shown in table XXVIII.

(U) Table XXVIII. Mechanical Gimbal Weights, lb

Components	Engine Size 250K	350K
Servoactuators (2)	90	102
Main Pump	16	22
Accumulator-Reservoir	75	88
Auxiliary Pump and Motor	20	25
Hydraulic Fluid	18	25
Plumbing	35	42
Brackets and Mounts	30	30
Electrical Cables	6	6
Total	290	340

(U) The weight values were determined by scaling actuators and pumps from available literature. The hydraulic fluid was determined by system volume requirements and the remaining items were scaled from the Advanced Saturn/AEB studies. The total TVC weights used in the installation studies are shown in table XXIX. These values do not include the gimbal and required engine beef-up because they are included in the parametric engine weights in sections.

(U) Table XXIX. TVC System Weights, lb

Case	250K	350K
1	1450	1360
2	290	340
3	1450	1360
4	2320	2040
5	580	340
6	290	340

APPENDIX I
VEHICLE APPLICATIONS PACKAGE

(U) The 250K vehicle applications package prepared by Aerospace Corporation and supplied to Pratt & Whitney Aircraft in support of the high performance rocket engine program is reproduced in this Appendix. The salient changes to the 250K information for the 350K application study are also presented.

A. VEHICLE APPLICATIONS PACKAGE, 250K MODULE

(U) The 250K vehicle applications package is reproduced on the following pages.

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VEHICLE APPLICATIONS PACKAGE

Data prepared by AFSSD/Aerospace Corporation, at the request of AFRPL, in support of the applications study portion of the High Performance Cryogenic Rocket Engine ADP.

Prepared by: C. H. Builder
W. A. Fey
M. G. Hinton
R. Krueger

Aerospace Corporation; El Segundo, California

Date: September 1965

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(C)(U) VEHICLE APPLICATIONS PACKAGE

(C)(U) TABLE OF CONTENTS

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1.0 (C)/U) INTRODUCTION

1.1 (U) Purpose

The purpose of the "Applications Package" is to provide a meaningful base upon which various High Performance Cryogenic Rocket Engine concepts may be compared. A consistent and workable set of ground rules has been established in this package; furthermore, a number of different vehicle concepts have been chosen to illustrate the diversity of potential applications.

The information and use instructions presented in this package have been formulated so that: (1) the contractor must face many vehicle related installation problems that would be involved in a real situation. To this end, considerable latitude is provided in the design of engine thrust structure, propellant ducting, and thrust vector control schemes; (2) the contractor is not overburdened by the vehicle design and performance analysis aspects of the overall problem, but yet can determine a meaningful performance index. Therefore, trajectory and trajectory loss data, along with appropriate instructions, are provided.

1.2 (C)(U) Outline of Package Contents

1.2.1 Vehicle Applications Cases (Section 2.0)

(C) Six different vehicle applications cases have been selected as being typical for the potential application of High Performance Cryogenic Rocket Engines. They are:

Case 1: Expendable First Stage Rocket

Case 2: Expendable Second Stage Rocket

Case 3: Expendable Single Stage to Orbit Rocket

Case 4: Recoverable First Stage (RTOHL)

Case 5: Recoverable Second Stage, Pickaback Configuration

Case 6: Recoverable Second Stage, Tandem Configuration

1.2.2 (U) Instructions are provided to explain the ground rules, the use of data presented, and the computation of performance indices. Included are:

Engine Installation Instructions

Fairing and Interstage Instructions

Instructions on Making Engine Performance Estimates

Instructions on Trajectory Integration

Instructions on the Calculation of Performance Index

1.2.3 (U) Technical Data Enclosures (Section 4.0)

Data necessary to perform the various computations, designs, etc., are included in this section, in the following order:

Installation Diagrams

Engine Installation Data

Velocity-Altitude Histories

Trajectory Loss Parameters

Burnout Weight Correction Factors

Performance Index Equation and Constants

1.3 (C)(U) General Ground Rules

The following ground rules and definitions are general and apply to all of the six vehicle cases defined in this package. Data pertaining to specific vehicles, as well as instructions on operational items and calculations, will be given later.

1.3.1 (C)(U) Engine Module Definition

(U) It is desired to use the same basic engine module in all six vehicle cases; however, since both first and second stage cases are considered, the utilization of an identical module, "across the board" is not realistic. In this Applications Package, the engine module is defined to be the upper stage module, developing 250,000 pounds (f) of thrust at vacuum conditions. The Contractor will select a single nozzle expansion ratio for use in the three upper stage applications (cases 2, 5, and 6), and will then determine engine flow rate.

This engine flow rate, as well as such parameters as chamber pressure, MUST remain constant for all six vehicle cases. The thrust of the engine modules to be used in the first stage application, will then be determined by the criteria in the next paragraph.

(U) In defining the first stage module, the Contractor should make every effort to minimize the differences between first stage and upper stage engine modules. It is anticipated that the eventual selection of the best engine concept will be made on the basis of single module versatility, as well as performance factors. Those Contractors proposing engine concepts using cylindrical or conical chambers with DeLaval nozzles should select appropriate nozzle expansion ratios for each of the three first stage cases. Module thrust levels will be determined using flow rate and chamber pressure values identical to the upper stage module. Those Contractors proposing engine concepts using annular

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chambers with aerodynamic nozzles should use the same chamber nozzle configuration selected for the upper stage, and determine first stage thrust levels by taking advantage of the altitude compensation features of aerodynamic nozzles. The Contractor's turbomachinery, turbine gas generation system, control system, and other module subcomponents MUST remain the same for all six vehicle cases. If subsequent optimization indicates to the Contractor that these ground rules are overly restrictive, he should feel free to define more suitable approaches in addition to those outlined above. Any differences in hardware between this more suitable approach, and the outlined approach must be identified in detail.

(C) In determining the major engine module parameters, a selection of mixture ratio is necessary. The basic engine module must be capable of operating between mixture ratios (oxidizer to fuel, by weight) of 5.0 and 7.0; in addition, the flow rate versus mixture ratio characteristics of the modules used in all six cases must be identical. The Contractor may find it necessary to perform several Performance Index iterations to optimize mixture ratio. A value of 6.0 is suggested as a nominal starting point.

1.3.2 (U) Responsibilities of the Contractor

The contractor is responsible for providing three principal items for each of the six vehicle configurations.

(a) A definition of the engine module configuration, along with all installation hardware (attachment structure, propellant ducting, thrust vector control system), as specified in Section 3.1.

(b) A definition of the geometry of interstages and fairings as specified in Section 3.2.

(c) A determination of the performance index, utilizing contractor-generated information from (a) and (b), in conjunction with data and instructions in this package.

1.3.3 (C) Contractor Objectives

The contractor's prime objective should be to define a single engine module which provides a high Performance Index to each of the six different vehicle applications with as few hardware differences as possible. Therefore, the contractor is encouraged to perform his own tradeoff studies so that his single module characteristics will result in a minimum of compromise for the six cases presented. In pursuing this prime objective, the Contractor should also have several corollary objectives, to wit:

(a) To consider all of the actual problems involved in the selection of an engine configuration, and to make realistic tradeoffs and compromises in arriving at final performance values, engine weights, and system configuration.

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(b) To consider all of the problems involved in adapting the basic module to different v' icle configurations, and to select intelligent design compromises.

(c) To remember that the eventual applications are advanced vehicles; and that these vehicles, whether recoverable or expendable will demand increased engine operational lifetimes and decreased engine maintenance requirements than now being demonstrated with existing engines.

1.3.4 (U) Interpretation of Results

The resultant figure of merit that the contractor will determine for each of the six applications cases is the "Performance Index." In actuality, the performance index is the stage burnout weight, less the weight of the engine installation and engine influenced hardware. This performance index IS NOT a payload value, and should not be misconstrued as such. It is merely a parameter which has been defined to reflect differences in advanced engine concepts (as well as variations of a given concept) for several typical Advanced Launch Vehicle stages. It is not to be used as a parameter to reflect differences in vehicle concepts.

Because of the nature of the Performance Index, the value determined for any given case IS NOT comparable between the other cases; due to differences in gross weight, structure factor, and so on. For a given case, however, the higher the Performance Index, the better.

2.0 (C)(U) CASE DESCRIPTIONS

2.1 (C)(C) Case Number 1, Expendable First Stage Rocket

2.1.1 (C) Application Description

Advanced Launch Vehicle Case #1 represents an expendable first stage of a vertically-launched, multistage launch vehicle. The stage is a cylindrical stage, 23 feet in diameter, with the liquid oxygen tank located forward of the liquid hydrogen tank. Engines and associated installation hardware are to be attached to a structural ring on the aft end of the stage. The oxygen propellant supply line(s) is located external to the vehicle, the hydrogen supply is centrally located at the bottom of the hydrogen tank.

2.1.2 (U) Stage Gross Weight

The gross weight of this stage is 800,000 pounds (m). For purposes of this exercise, the gross weight includes all weights above the stage. The gross weight and lift-off weight are assumed to be the same.

2.1.3 (U) Engine Modules and Nominal Stage Thrust

The total thrust of this stage shall be that thrust provided by five (5) engine modules. The individual engine module thrust (at an ambient pressure of 14.7 psia) is to be determined by the contractor in accordance with instructions in Section 1.3.1, with the restriction that the thrust to gross weight ratio of the stage at lift off must exceed 1.20.

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2.1.4 (U) Stage Velocity Increment

The stage burnout velocity is 10,000 ft/sec, starting from an initial velocity of zero.

2.1.5 (U) Anticipated Issues

It is anticipated that the following issues will be important in adapting the engine module to this vehicle case:

(a) Well organized multiple engine installation, providing low overall weight, good accessibility, minimum pre-flight maintenance

(b) Vehicle Thrust Vector Control

2.2 (C)(C) Case Number 2, Expendable Second Stage Rocket

2.2.1 (C) Application Description

Advanced Launch Vehicle Case #2 represents an expendable second stage of a vertically-launched, multistage launch vehicle. The stage is a cylindrical stage, 17 feet in diameter, with the liquid hydrogen tank located forward of the liquid oxygen tank. The engine is to be attached to the main structural ring with an appropriate load-carrying frame. A single oxygen supply duct is centrally located at the bottom of the oxygen tank, and a single hydrogen supply duct is available anywhere within the inside chamber of the main structural ring.

2.2.2 (U) Stage Gross Weight

The gross weight of this stage is 250,000 pounds (m). For purposes of this exercise, the gross weight includes all weights above the stage. The gross weight and ignition weight are assumed to be the same.

2.2.3 (U) Engine Module and Nominal Stage Thrust

The total thrust of this stage shall be that thrust provided by one (1) engine module. The individual engine module thrust shall be 250,000 pounds (f) at vacuum conditions.

2.2.4 (U) Stage Velocity Increment

The stage burnout velocity is 26,000 ft/sec starting with an initial velocity of 10,000 feet/sec.

2.2.5 (U) Anticipated Issues

It is anticipated that the following issues will be important in adapting the engine module to this vehicle case:

(a) High Specific Impulse

(b) Restart Capability

2.3 (C)(C) Case Number 3, Expendable Single Stage to Orbit Rocket

2.3.1 (C) Application Description

Advanced Launch Vehicle Case #3 is a vertically-launched, expendable, single stage to orbit vehicle. The vehicle is cylindrical with a diameter of 26 feet, and the oxygen tank is located forward of the hydrogen tank. The oxygen supply lines extend through tunnels in the hydrogen tank to the vehicle engine interface, whereas a single hydrogen supply line is located at the bottom of the hydrogen tank. A main structural ring is used as the attach point for the engine/thrust mount assemblies.

2.3.2 (U) Stage Gross Weight

The gross weight of this stage is 800,000 pounds (m). For purposes of this exercise, the gross weight includes all weights above the stage; furthermore, the gross weight and lift-off weight are assumed to be the same.

2.3.3 (U) Engine Modules and Nominal Stage Thrust

The total thrust of this stage shall be that thrust provided by five (5) engine modules. The individual engine module thrust (at an ambient pressure of 14.7 psia) is to be determined by the contractor in accordance with instructions in Section 1.3.1, with the restriction that the thrust to gross weight ratio of the stage at lift off exceeds 1.20. Capability for thrust variation will be required to limit the stage acceleration to a maximum of 5.0 g's.

2.3.4 (U) Stage Velocity Increment

The stage burnout velocity is 26,000 f/sec, starting with an initial velocity of zero.

2.3.5 (C) Anticipated Issues

It is anticipated that the following issues will be important in adapting the engine module to this vehicle case:

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(a) Well organized multiple engine installation providing low overall weight, good accessibility, minimum pre-flight maintenance

(b) Compromise between sea level and altitude performance

(c) Vehicle Thrust Vector Control

(d) Engine Module Throttling

2.4 (C)(C) Case Number 4, Recoverable First Stage Rocket (VTOHL)

2.4.1 Application Description

(C) Advanced Launch Vehicle Case #4 is a winged, recoverable, first stage (VTOHL) of a multistage launch vehicle. The stage is cylindrical in body form (22.5 ft diameter) with wings extending therefrom. The oxygen tank is located forward of the hydrogen tank. Multiple oxygen supply lines pass through tunnels in the hydrogen tank. A structural ring, at the aft end of the stage, is provided for accepting the engine thrust loads.

2.4.2 (U) Stage Gross Weight

The gross weight of this stage is 1,280,000 pounds (m). For purposes of this exercise, the gross weight includes all weights above the stage; furthermore, the gross weight and lift-off weight are assumed to be the same.

2.4.3 (U) Engine Modules and Nominal Stage Thrust

The total thrust of this stage shall be that thrust provided by eight (8) engine modules. The individual engine module thrust (at an ambient pressure of 14.7 psia) shall be determined by the contractor in accordance with instructions in Section 1.3.1; with the restriction that the thrust to gross weight ratio of the stage at lift off must exceed 1.20.

2.4.4 (U) Stage Velocity Increment

The stage burnout velocity is 10,000 f/sec, starting with an initial velocity of zero.

2.4.5 Anticipated Issues

(C) It is anticipated that the following issues will be important in adapting the engine module to this vehicle case:

(a) Well organized multiple engine installation, providing low overall weight, good accessibility, minimum maintenance

(b) Compact engine installation, i.e., minimum volume

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(c) True engine reusability

(d) Vehicle Thrust Vector Control

2.5 (C)(C) Case Number 5, Recoverable Second Stage Rocket, Pickaback Configuration

2.5.1 Application Description

(C) Advanced Launch Vehicle Case #5 is a recoverable second stage of a multistage vehicle configuration. This stage is boosted to near vacuum conditions by a recoverable first stage in a pickaback configuration. The second stage is of the lifting body design, with non-integral cylindrical propellant tanks (one oxygen and two hydrogen tanks) arranged in a side-by-side manner. A structural frame is provided at the aft end to accommodate the engine installation. Each propellant tank has a single discharge outlet, located at the bottom of each tank.

2.5.2 Stage Gross Weight

(U) The gross weight of this stage is 500,000 pounds (m). For purposes of this exercise, the gross weight includes all weights above the stage; furthermore, the gross weight and ignition weight are assumed to be the same.

2.5.3 Engine Module and Nominal Stage Thrust

(U) The total thrust of this stage shall be that thrust provided by two (2) engine modules. The individual engine module thrust shall be 250,000 pounds (f) at vacuum conditions.

2.5.4 Stage Velocity Increment

(U) The stage burnout velocity is 26,000 ft/sec, starting with an initial velocity of 10,000 ft/sec.

2.5.5 Anticipated Issues

(C) It is anticipated that the following issues will be important in adapting the engine module to this vehicle case:

(a) High Specific Impulse

(b) Compact Engine Installation

(c) Vehicle Thrust Vector Control

(d) Restart Capability

(e) True engine reusability

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2.6 (C)(C) Case Number 6, Recoverable Second Stage Rocket, Tandem Configuration

2.6.1 Application Description

(C) Advanced Launch Vehicle Case #6 is a recoverable second stage of a multistage vehicle configuration. This stage is boosted to near vacuum conditions by a rocket booster in a conventional tandem configuration. The second stage is of the lifting body type, with integral propellant tanks. A Structural aft frame, with four (4) mounting pads, is provided for attaching the engine to the vehicle. Single oxygen and hydrogen outlet line locations are specified in the base of the vehicle.

2.6.2 Stage Gross Weight.

(U) The gross weight of this stage is 250,000 pounds (m). For purposes of this exercise, the gross weight includes all weights above the stage; furthermore, the gross weight and ignition weight are assumed to be the same.

2.6.3 Engine Module and Nominal Stage Thrust

(U) The total thrust of this stage shall be that thrust provided by one (1) engine module. The individual engine module thrust shall be 250,000 pounds (f) at vacuum conditions.

2.6.4 Stage Velocity Increment

(U) The stage burnout velocity is 26,000 ft/sec starting with an initial velocity of 10,000 ft/sec.

2.6.5 Anticipated Issues

(C) It is anticipated that the following issues will be important in adapting the engine module for this vehicle case:

(a) High Specific Impulse

(b) Restart Capability

(c) True Engine Reusability

3.0 (U) INSTRUCTIONS

3.1 Engine and Engine Installation Instructions

3.1.1 Engine System Instructions

Prior to the determination of the Performance Index, the contractor will find it necessary to determine the exact thrust, specific impulse, and mixture ratio of his engine module, along with weights and geometry of the

entire module and installation. In specifying the module design, the contractor should consider all of the necessary facets of engine operation, vehicle installation, checkout, and maintenance that would be part of an operational, manned, vehicle program.

The contractor's engine module must conform to the following:

a. Vehicle Propellant Interface Conditions

The conditions of the propellants available for the engine will (in reality) vary considerably among the six different applications cases. In order to simplify the problem, a typical set of conditions has been postulated as being representative of all vehicle cases. These conditions are specified in Section 4.2.1. The contractor is required to specify a module design which can operate satisfactorily with these inlet conditions both during steady state and transient engine operation.

b. Thrust Vector Control Requirements

The engine modules utilized in the various cases must be capable of providing Thrust Vector Control (TVC). The TVC requirements are specified in Section 4.2.2. The contractor has the option of selecting either mechanical movement of the thrust vector, or secondary fluid injection as the scheme by which to achieve the required TVC. Once a scheme is selected, the contractor is responsible for defining all of the subcomponents of the TVC system in order to arrive at a realistic weight estimate.

In this Applications Package, other vehicle interface problems such as tank pressurization requirements, guidance and steering interfaces, telemetry, etc., are not to be considered in detail. However, the contractor will be required to make reasonable estimates on weights of such items as malfunction detection/instrumentation; this will be discussed further in Section 3.1.3.

3.1.2 Use of Installation Diagrams

An installation diagram is provided for each of the six cases in Section 4.1. These diagrams depict the aft end of a vehicle that is typical of the various cases defined. Included in the diagrams are aft end vehicle dimensions, point(s) at which engine thrust structure is mated to the vehicle, location(s) of propellant outlet duct(s), and other pertinent information. The contractor is responsible for defining ALL of the installation paraphernalia that is required to adapt the engine or engines to the interfaces defined in the diagrams; some of the major items are:

a. Thrust Structure

The contractor is required to define a suitable thrust structure that will carry the thrust loads to the defined vehicle location. Design parameters, dimensions, and weights must be specified. The structure shall withstand all of the forces resulting from the thrust of the engine module(s) during transient

and steady state operation. For design purposes, the ultimate strength shall provide for a minimum of 1.50 times the maximum loading condition. The thermal environment of structural members will be discussed in Section 3.1.2(d).

b. Propellant Feed Lines

The contractor is required to define the feed lines which will carry propellants from the defined interfaces to the engine(s). In some cases, the contractor must choose the number of feed lines from the tank, and specify their location. The sizes of lines, design details, and weights must be provided.

c. Thrust Vector Control System

The Contractor is required to provide a suitable thrust vector control system: details of this system have been discussed in Section 3.1.1 All hardware necessary for the system and its installation must be specified.

d. Module and Installation Environment

The entire engine module installation must operate satisfactorily, at ambient pressure between sea level and vacuum conditions. It must be able to start, run and stop properly throughout any flight path; furthermore, the engine and installation hardware must withstand all vibration and acceleration criteria encountered during normal usage. Prior to the entry of propellants, a -60°F to +160°F environment should not have any detrimental effects on the engine system. It is to be assumed that the maximum temperature of components and hardware not in contact with propellants or turbine gases will be limited to 500°F during the vehicle flight. Material choices, insulation, loading limits, etc., are to be compatible with these environmental constraints.

3.1.3 Engine Installation Weights

The computation of a meaningful performance index requires a rigorous bookkeeping procedure for engine module weights and installation hardware weights. In order that consistency can be maintained among the various cases, some representative weight breakdowns are furnished in Section 4.2.3 for the module and for the overall installation. Although the contractor is not limited to this breakdown, all items specified must be included.

The contractor is encouraged to use his best judgment in identifying those items where insufficient vehicle data is available to allow detailed weight estimates. Items which fall into this category are heat exchangers for pressurization, engine component heat protection, malfunction detection systems, etc.

3.2 Fairing and Interstage Instructions

3.2.1 Instructions for Calculating Fairing Areas

Most of the applications cases require the layout of an aerodynamic fairing in the vicinity of the engine compartment. In some of the cases, the fairing is required to enclose the entire engine compartment, encapsulating

the engine. No structural design of these fairings is required. The fairing layouts should be considered as nothing more than an artificiality introduced into the calculations as a means for assessing certain weights associated with the geometry of the engine installation. The fairing layouts will be used by the contractor to determine an area term, A_f, which is required in calculating the performance index, W_x, for the propulsion system. Furthermore, these fairing dimensions are required to determine a suitable correction factor to account for engine installation drag effects.

General

a. Fairings shall consist of skirts and closures.

b. Skirts shall extend from the vehicle body surface(s) aft to the plane of the engine nozzle throat.

c. Closures shall extend from the skirt surface(s) aft until the base of the vehicle is closed.

d. Skirts shall not interfere with or be penetrated by any part of the engine installation, even at maximum thrust deflection position.

e. Closures shall not be penetrated by any part of the engine installation when the engine is at a zero thrust deflection position.

f. Skirts shall consist of cylindrical, conical, or planar surfaces.

g. Closures shall consist of cylindrical, conical, planar, or ogival surfaces.

h. Ogival surfaces shall be generated by a circular arc which

1. is tangent to the cylindrical, conical, or planar surface at its forward end and

2. terminates on the longitudinal axis of the vehicle, and

3. has a radius three times the perpendicular distance from the point of tangency to the longitudinal axis of the vehicle.

i. The angle of all conical or planar surfaces with respect to the longitudinal axis of the vehicle shall not exceed fifteen (15) degrees.

j. The fairing area, A_f, shall be the sum of the surface areas of the skirt and the closure, in square feet.

Case 1

A skirt shall be provided. No closure is required. The skirt shall be made up of conical and/or cylindrical surfaces such that any cross-section of the skirt in a plane perpendicular to the longitudinal axis of the vehicle is a circle. The forward end of the skirt must be attached to the vehicle body surface at some point between the main structural ring and seventy-five (75) inches forward of the main structural ring.

Case 2

No fairing is required. The fairing area, A_f, shall be taken as zero.

Case 3

A skirt shall be provided. No closure is required. The skirt shall be made up of conical and/or cylindrical surfaces such that any cross-section of the skirt in a plane perpendicular to the longitudinal axis of the vehicle is a circle. The forward end of the skirt must be attached to the vehicle body surface at some point between the main structural ring and one-hundred-ten (110) inches forward of the main structural ring.

Case 4

A skirt and closure shall be provided. The skirt shall be made up of conical and/or cylindrical surfaces. The closure shall be made up of conical and/or cylindrical surfaces plus an ogive. All surfaces for both skirt and closure shall be such that any cross-section in a plane perpendicular to the longitudinal axis of the vehicle is a circle. The forward end of the skirt must be attached to the vehicle body surface at some point between the main structural ring and fifty (50) inches forward of the main structural ring.

Case 5

A skirt and closure shall be provided. The skirt shall be made up of planar surfaces only. The forward end of the skirt must be attached to the vehicle body surfaces at the main aft frame of the vehicle. The closure shall be made up of planar surfaces and a two-dimensional ogive. The surfaces of the skirt and closure shall be such that any cross-section perpendicular to the longitudinal axis of the vehicle is a rectangle. The major dimension (longer side) of this rectangle shall be constant, forty-five (45) feet long.

Case 6

A skirt shall be provided. No closure is required. The skirt shall be made up of planar surfaces only. The forward end of the skirt must be attached to the vehicle body surfaces at the main aft frame of the vehicle. The surfaces of the skirt shall be such that any cross-section perpendicular to the longitudinal axis of the vehicle is a hexagon geometrically proportional or similar to the main aft frame of the vehicle.

3.2.2 Instructions for Calculating Interstage Areas

Two of the applications cases require the layout of interstages in the vicinity of the engine compartment. No structural design of these interstages is required. The layouts are intended as a means for assessing certain weights which may be associated with the geometry of the engine installation. The interstage layouts will be used by the contractor to determine an area term, A_i, which is required in calculating the Performance Index, W_x, for the propulsion system.

General

a. Interstages shall not be penetrated by any part of the engine installations when the engine is at a zero thrust deflection position.

b. The interstage area, A_i, shall be the surface area of the interstage in square feet.

Cases 1, 3, 4, and 5

No interstage is required. The interstage area, A_i, shall be taken as zero.

Case 2

The interstage shall be conical surface having a diameter at the forward end of seventeen (17) feet and a diameter at the aft end of twenty-three (23) feet. The forward end of the interstage shall be attached to the main structural ring. The interstage shall have a minimum length, measured along the longitudinal axis of the vehicle, of eight (8) feet, but shall be extended, as necessary, to the aftermost plane of any engine part.

Case 6

The interstage shall be a surface generated by the shortest lines drawn between the aft end of the skirt and a twenty (20) foot diameter circle whose plane is perpendicular to the longitudinal axis of the vehicle. No generating line element shall be inclined more than fifteen (15) degrees with respect to the longitudinal axis of the vehicle. The interstage shall be extended, as necessary, to the aftermost plane of any engine part.

3.3 Engine Performance Estimates

A velocity-altitude history is postulated for each application for the purpose of making engine performance estimates. Curves of this history are provided for Cases 1, 3, and 4 in Section 4.5. For Cases 2, 5, and 6 it shall be assumed that the engine operation occurs under vacuum conditions.

The ARDC 1959 atmosphere is to be used with the velocity-altitude histories to determine engine performance as a function of velocity. Thrust and specific impulse as a function of velocity are to be determined under the assumption that the thrust chambers are aligned with the vehicle axis, i.e., the cant angle is zero. Any effects of thrust vectoring are to be neglected for this computation. No maximum acceleration limits are applicable except to Case 3 where the instantaneous values of F/W shall not exceed 5. Thrust must be controlled in some manner (e.g., throttling, or shutting off modules) to remain within this limit.

3.4 Instructions for Trajectory Performance Integration

Trajectory performance for each applications case shall be computed by means of a modified form of the ideal rocket equation. The integral form of this equation is

$$\int_{W_o}^{W_b} \frac{dW}{W} = - \int_{V_o}^{V_b} \frac{dV}{I_s \, g_c (1 - \frac{W}{F} L)}$$

where W = the instantaneous weight of the vehicle, lb_m

W_o = the initial or gross weight of the vehicle, lb_m

W_b = the burnout weight of the vehicle, lb_m

V = the instantaneous velocity of the vehicle, ft/sec (non-rotating earth)

V_o = the initial velocity of the vehicle, ft/sec

V_b = the burnout velocity of the vehicle, ft/sec

I_s = the instantaneous specific impulse of the installed propulsion system, sec lbf/lbm

g_c = 32.174 ft/sec^2

F = the instantaneous thrust of the installed propulsion system, lbf

L = the instantaneous value of the trajectory loss parameter, dimensionless

Direct integration of this equation is not possible because of the W under the velocity integral. Therefore, integration shall be carried out in a step-wise manner using the following algebraic equation for each step:

$$\Delta W = - \frac{W \, \Delta V}{I_s \, g_c (1 - \frac{W}{F} L)}$$

where ΔW = the weight change of the vehicle during each step, lbm

ΔV = the velocity change of the vehicle during each step, ft/sec

The velocity increment, ΔV, shall be taken at 200 ft/sec for each step. The other terms (W, I_s, F, and L) shall be evaluated at the beginning of each step. In this manner, a weight history of the vehicle with velocity (W - V diagram) shall be calculated from the initial to the burnout conditions for the vehicle. The initial and burnout velocities have been previously specified in Section 2.0.

The burnout weight of the vehicle, W_b, shall be corrected for fairing drag by the expression:

$$W_{bc} = CW_b$$

where W_{bc} = the corrected burnout weight, lbm

C = the stage burnout weight correction factor

Curves for C as a function of maximum fairing dimensions are furnished. For Cases 2 and 6, C = 1.0 for any configuration

3.5 Instructions on the Calculation of the Performance Index

The contractor will compute a "Performance Index" for each of the six vehicle cases. This performance index is to be computed using the algebraic equation specified in Section 4.6, along with the specified constants and contractor computed or supplied variables. The value of the performance index which is determined for each vehcile case will be the figure of merit for that case. As stated earlier, the figures of merit are NOT comparable between cases, but are comparable for each case. Consequently, the contractor may wish to optimize the performance index for each individual case, in terms of mixture ratio, engine configuration, etc.

4.0 TECHNICAL DATA ENCLOSURES

4.1 Installation Diagrams

4.1.1 Installation Diagram, Case #1

4.1.2 Installation Diagram, Case #2

4.1.3 Installation Diagram, Case #3

4.1.4 Installation Diagram, Case #4

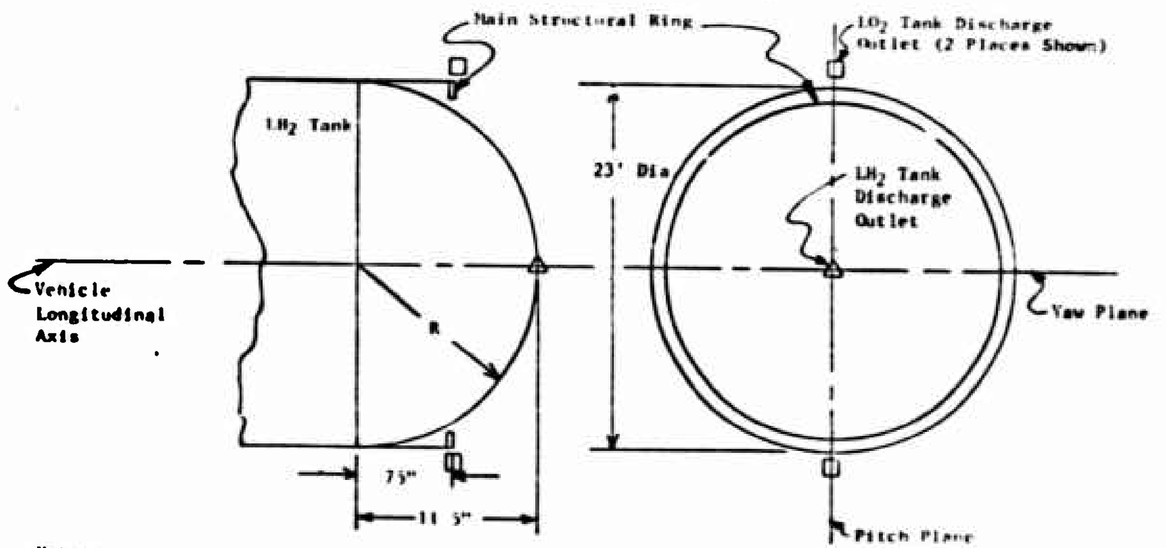
4.1.5 Installation Diagram, Case #5

4.1.6 Installation Diagram, Case #6

4.1.1 Installation Drawing, Case 1

CASE I

Gross Takeoff Weight = 800,000 lbs.
Five (5) Engine Modules



Notes:

(1) Thrust Mount Structure Shall be Attached to Main Structural Ring (No. of Points & Location to be Determined by Contractor).

(2) Up to Five (5) LO_2 Tank Discharge Outlets May be Utilized by Contractor (Location to be Symmetrical)

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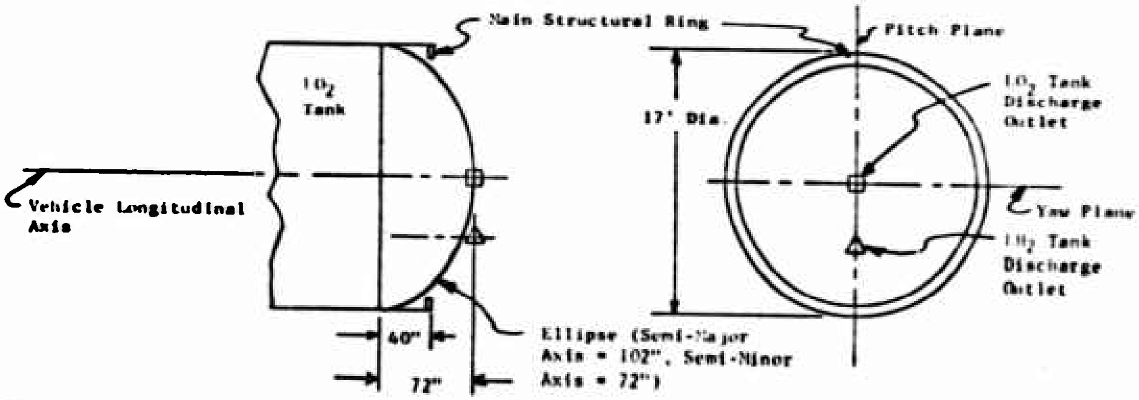
4.1.2 Installation Diagram, Case 2

CASE 2

Stage Ignition Weight = 250,000 lbs.
One (1) Upper Stage Engine Module @ F_v = 250,000 lbs.



Notes:

(1) Thrust Mount Structure Shall be Attached to Main Structural Ring (No. of Points & Location to be Determined by Contractor).

(2) Radial & Angular Location of LH_2 Tank Discharge Outlet Optional Within Constraints of Main Structural Ring Diameter.

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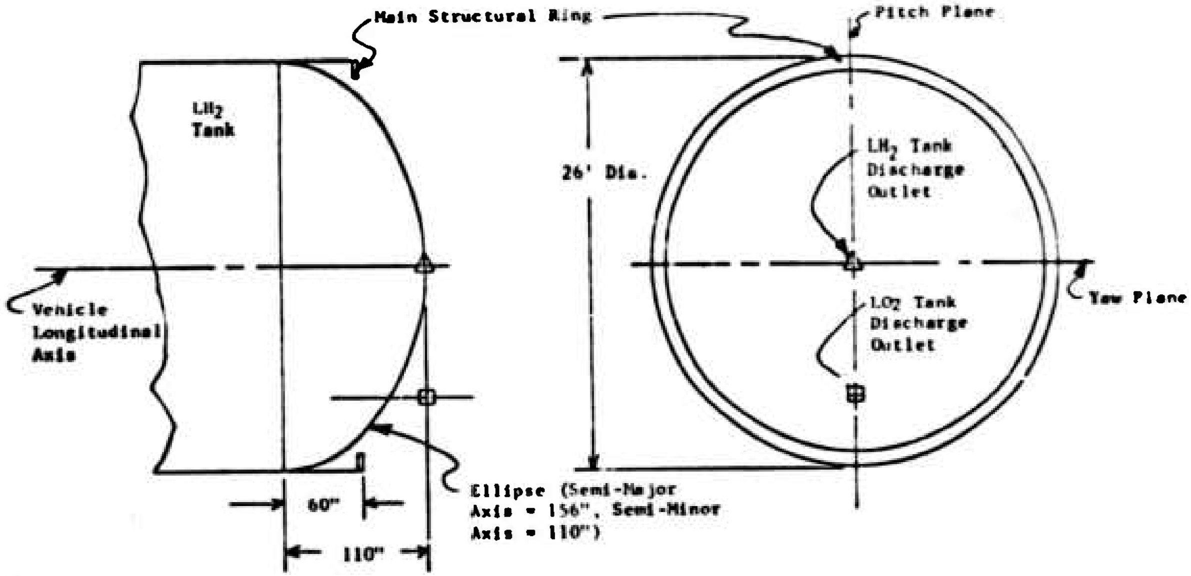
4.1.3 Installation Diagram, Case 3

CASE 3

Gross Takeoff Weight = 800,000 lbs.
Five (5) Engine Modules



Notes:

(1) Thrust Mount Structure shall be Attached to Main Structural Ring (No. of Points & Location to be Determined by Contractor).

(2) Up to Five (5) LO_2 Tank Discharge Outlets May be Utilized by Contractor (Locations to be Symmetrical). Radial Location Optional Within Constraints of Main Structural Ring Diameter.

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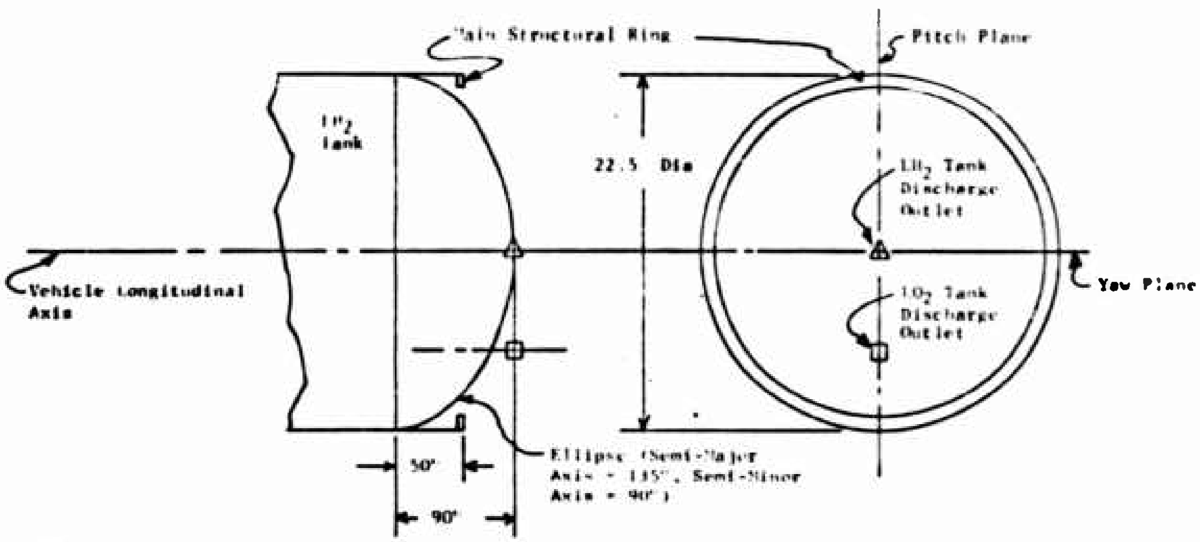
4.1.4 Installation Diagram, Case 4

CASE 4

Gross Takeoff Weight = 1,280,000 lbs.
Eight (8) Engine Modules



Notes:

(1) Thrust Mount Structure Shall be Attached to Main Structural Ring (No. of Points & Location to be Determined by Contractor).

(2) Up to Four (4) LO_2 Tank Discharge Outlets May be Utilized by Contractor (Locations to be Symmetrical) Radial Location Optional Within Constraints of Main Structural Ring Diameter.

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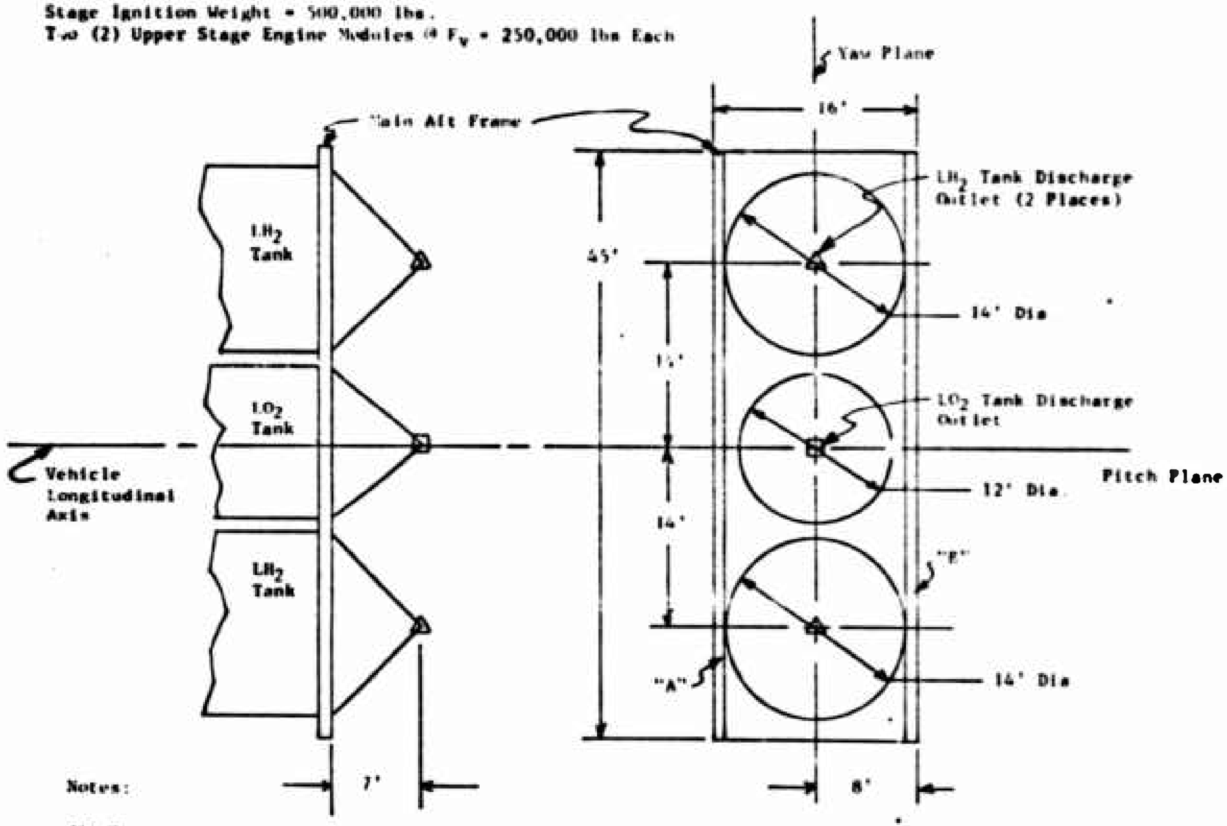
4.1.5 Installation Diagram, Case 5

CASE 5

Stage Ignition Weight = 500,000 lbs.
Two (2) Upper Stage Engine Modules @ F_v = 250,000 lbs Each



Notes:

(1) Thrust Mount Structure Shall be Attached to Members "A" & "B" of Main Aft Frame. No. of Points & Location to be Determined by Contractor.

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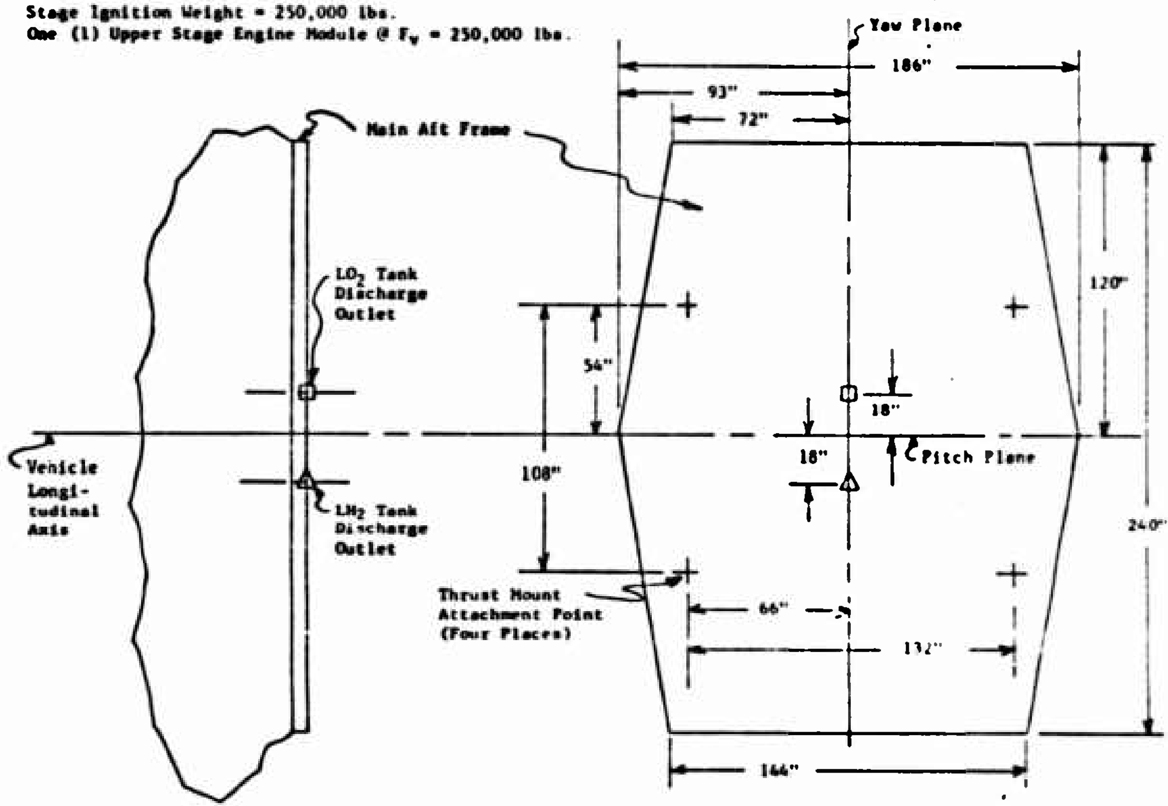
4.1.6 Installation Diagram, Case 6

CASE 6

Stage Ignition Weight = 250,000 lbs.
One (1) Upper Stage Engine Module @ F_v = 250,000 lbs.



Notes:

(1) Thrust Mount Structure Shall be Attached to Main Aft Frame at Four (4) Points as Indicated.

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4.2 Engine Installation Data

4.2.1 Vehicle/Engine Interface Propellant Conditions

4.2.2 Thrust Vector Control Requirements

4.2.3 Typical Engine Module and Installation Weight Formats

4.2.1 VEHICLE/ENGINE INTERFACE PROPELLANT CONDITIONS

(Conditions apply to all six vehicle cases)

A. PROPELLANT CONDITIONS

	Oxygen	Hydrogen
Total Pressure, psia	40	35
Propellant Temperature, ^oR	175.6	41.3
Propellant Density, lbm/ft^3	68.9	4.21
Propellant Vapor Pressure, psia	30	30

B. SUCTION HEAD AVAILABLE

The minimum Net Positive Suction Head (NPSH) AVAILABLE at the vehicle/engine interface is:

	Oxygen	Hydrogen
NPSH, available, feet	16	60

The engine module NPSH requirements, during start transients and steady state conditions, must be less than or equal to these available values.

4.2.2 THRUST VECTOR CONTROL REQUIREMENTS

TVC Scheme / Vehicles	Mechanical Movement (Hinging or Gimballing)	Secondary Fluid Injection
Case #1 Expendable First Stage Case #3 Expendable SSTO Case #4 Recoverable First Stage	Maximum Side Force / Maximum Axial Thrust* = 0.12	Maximum Side Force / Maximum Axial Thrust* = 0.07
	Total Impulse, Side / Total Impulse, Axial* = 0.02	Total Impulse, Side / Total Impulse, Axial* = 0.02
These three cases require pitch, yaw, and roll axis control		Average Side Force / Maximum Axial Thrust* = 0.035
Case #2 Expendable Second Stage Case #5 Recoverable Second Stage Case #6 Recoverable Second Stage	Maximum Side Force / Maximum Axial Thrust* = 0.12	Maximum Side Force / Maximum Axial Thrust* = 0.07
	Total Impulse, Side / Total Impulse, Axial* = 0.01	Total Impulse, Side / Total Impulse, Axial* = 0.01
These three cases require only pitch and yaw axis control.		Average Side Force / Maximum Axial Thrust* = 0.02

* Values taken in vacuum, based on total vehicle thrust

Data courtesy of C. Speisman, Propulsion Department, Aerospace

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4.2.3 Engine Module and Installation Weight Breakdown Format

The Contractor is required to itemize engine module and installation hardware weights. The following format is suggested to insure that all items are adequately specified when the total engine installation weight is predicted for the Performance Index computation. It is recognized that different engine concepts will not utilize all of the subcomponents listed below. As previously stated, the Contractor should make reasonable estimates for those items he is not specifically instructed to design.

(a) Engine Module Weights

Combustion Chamber Assembly

- Primary Combustion Chamber (Preburner)
- Primary Combustion Chamber Injector
- Secondary Combustion Chamber
- Secondary Combustion Chamber Injector
- Nozzle Extension(s)
- Aerodynamic Nozzle Centerbody and Shrouds

Oxidizer Turbopump Assembly

- Basic Pump/Turbine Assembly
- Mounting Structures Provisions
- Boost Pump or Separate Inducer

Fuel Turbopump Assembly

- Basic Pump/Turbine Assembly
- Mounting Structure and Provisions
- Boost Pump or Separate Inducer

Oxidizer Feed System (High Pressure)

Fuel Feed System (High Pressure)

Gas Generator Installation

Control System and Valves

Ignition System

Propellant Utilization System

Start System

Other Structure and Interconnects

(b) Engine Module Installation Weights

Thrust Structure

Oxidizer Feed System (Low Pressure)

Heat Protection System, Insulation

Instrumentation and Malfunction Detection System

Pressurization Hardware, Heat Exchangers

Thrust Vector Control System, Mechanical

Gimbal or Hinge Bearing and Structure

Actuators

Power Supply or Actuation Power Source

Additional Structure, Outriggers, etc.

Thrust Vector Control System, Secondary Injection

Manifolds, Lines and Plumbing

Injectant Valves

Structure, Nozzle Supports, etc.

4.3 Velocity-Altitude Histories*

4.3.1 Velocity-Altitude History, Case #1

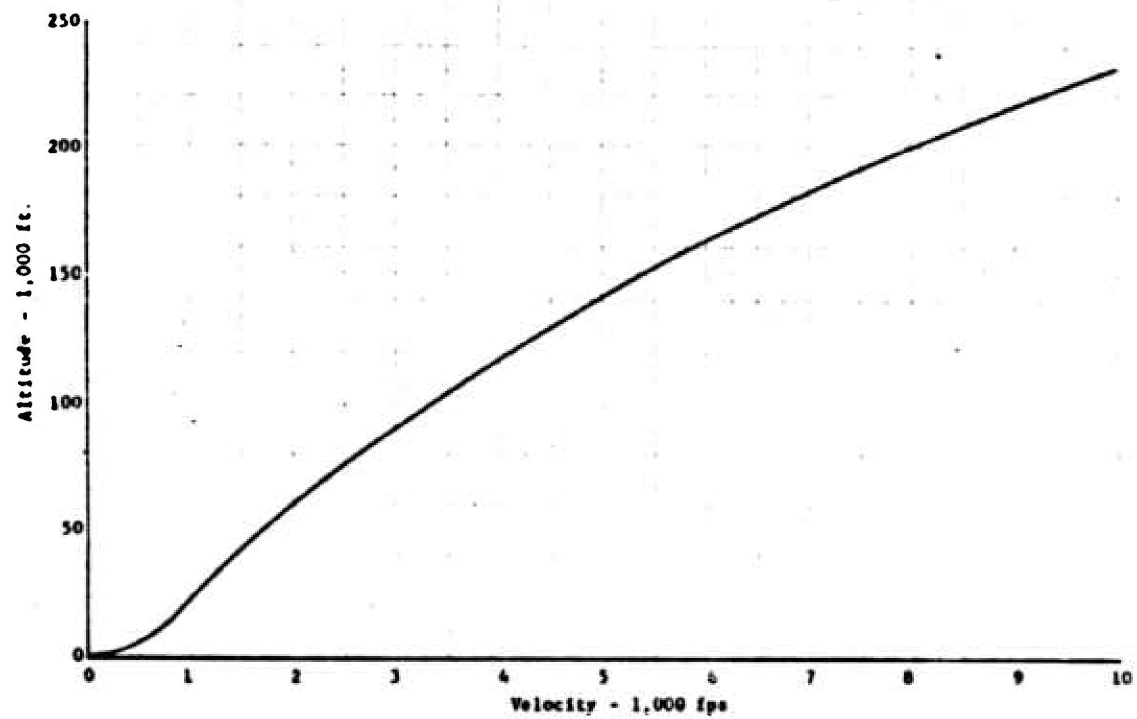
4.3.2 Velocity-Altitude History, Case #3

4.3.3 Velocity-Altitude History, Case #4

* No velocity-altitude curves needed for Cases #2, #5, and #6; vacuum conditions shall be assumed.

VELOCITY - ALTITUDE HISTORY - CASE 1

4.3.1 Velocity-Altitude History, Case 1



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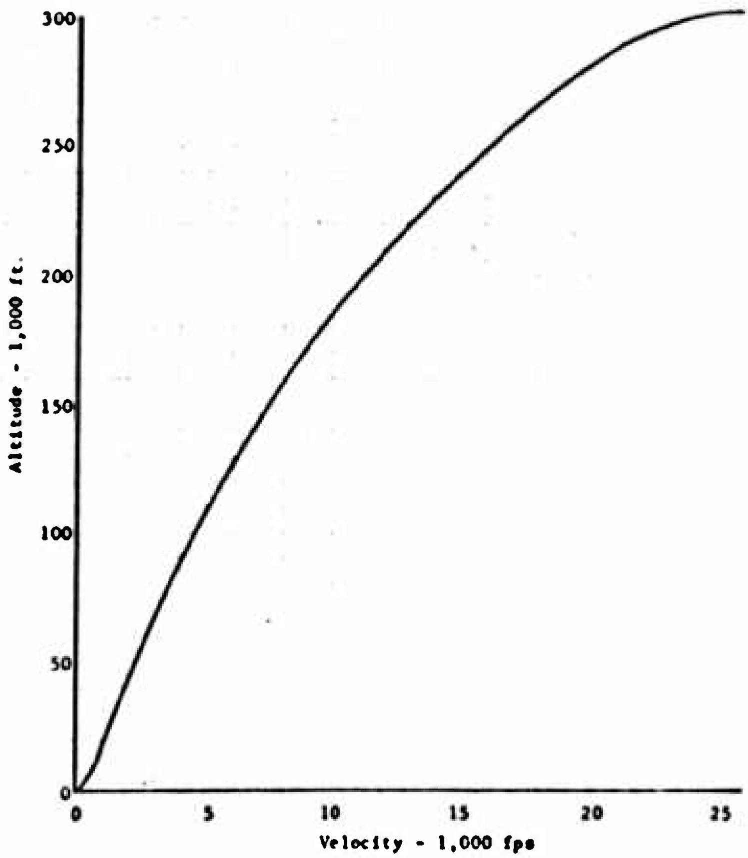
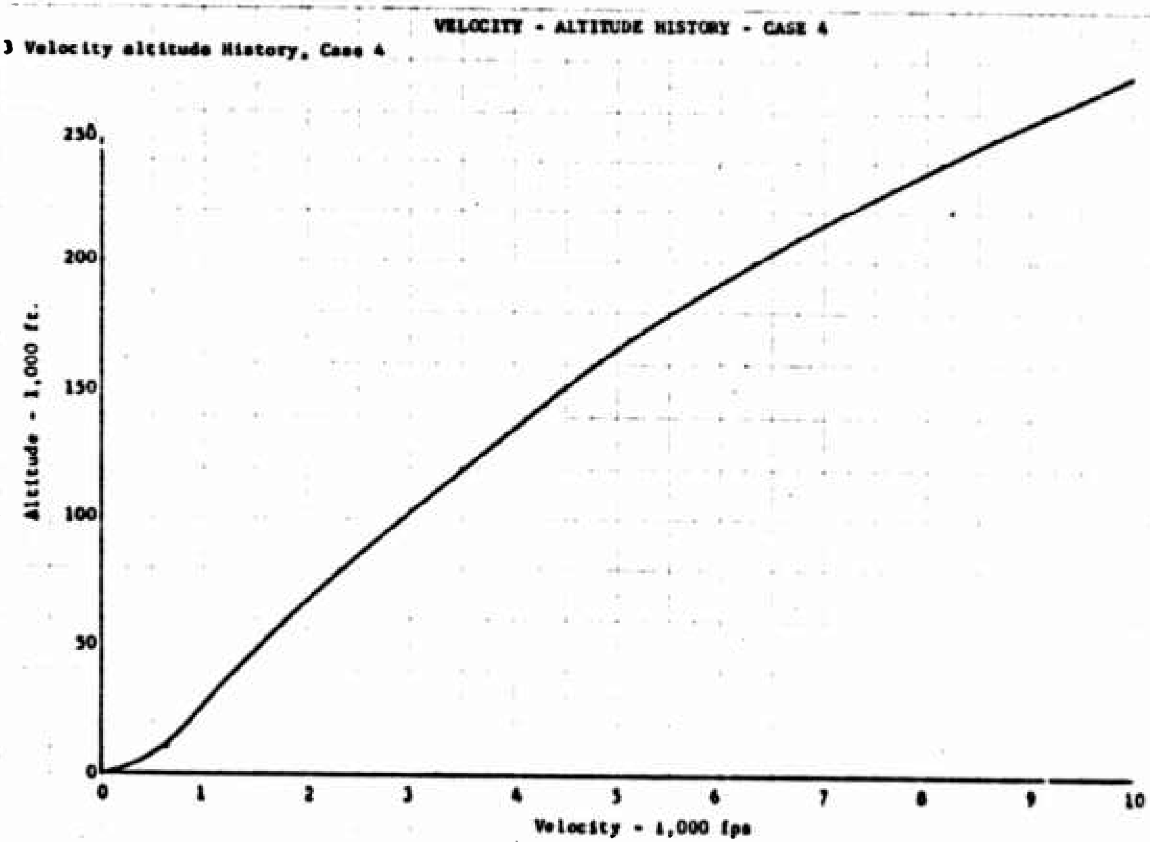
VELOCITY - ALTITUDE HISTORY - CASE 3

4.3.2 Velocity - Altitude History, Case 3



4.3.3 Velocity altitude History, Case 4



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4.4 Trajectory Loss Parameters

4.4.1 Trajectory Loss Parameter, Case #1

4.4.2 Trajectory Loss Parameter, Case #2

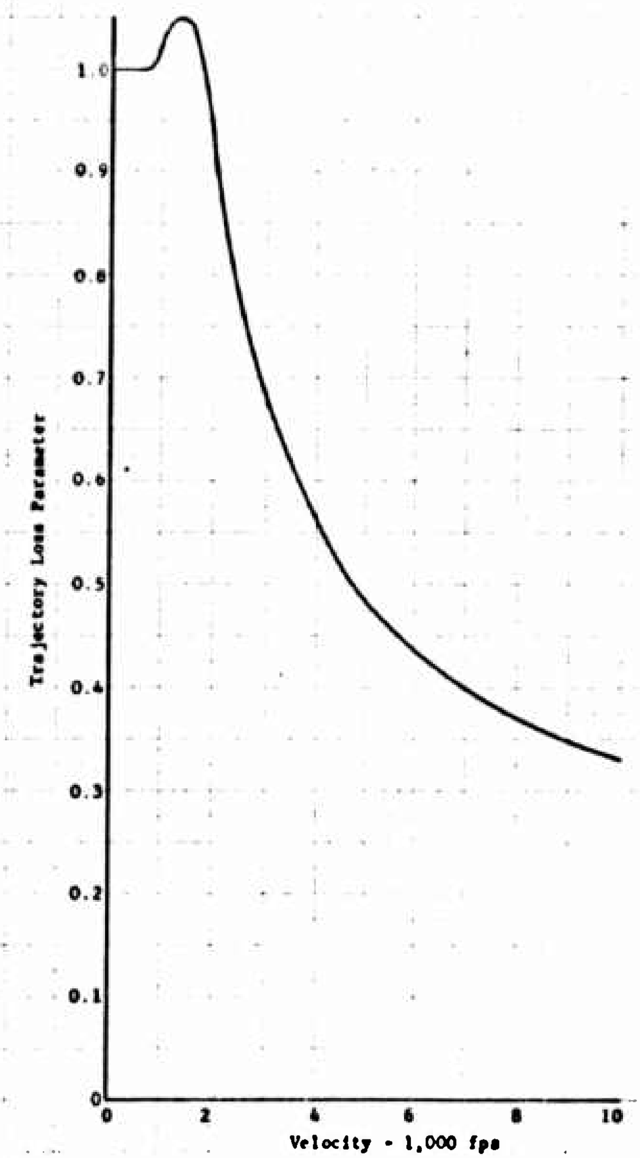
4.4.3 Trajectory Loss Parameter, Case #3

4.4.4 Trajectory Loss Parameter, Case #4

4.4.5 Trajectory Loss Parameters, Cases #5 and #6

TRAJECTORY LOSS PARAMETER - CASE 1

4.4.1 Trajectory Loss Parameter, Case 1



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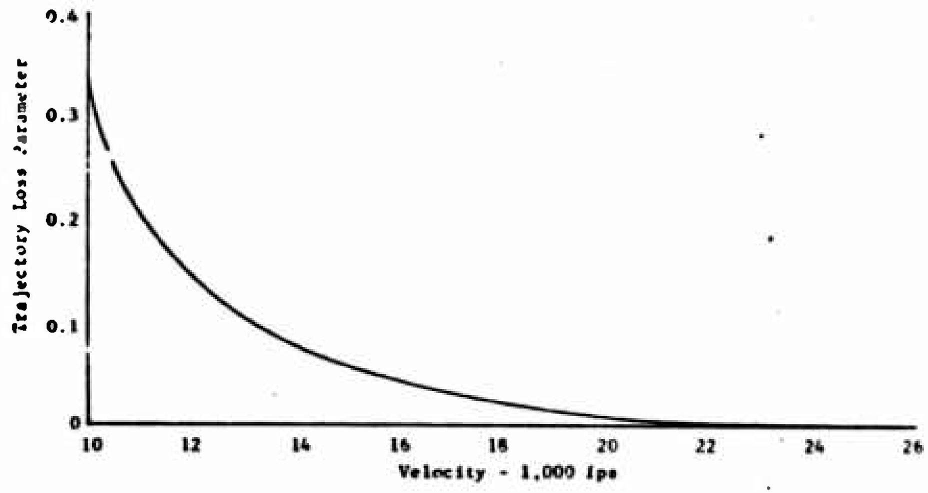
4.4.2 Trajectory Loss Parameter, Case 2

TRAJECTORY LOSS PARAMETER - CASE 2



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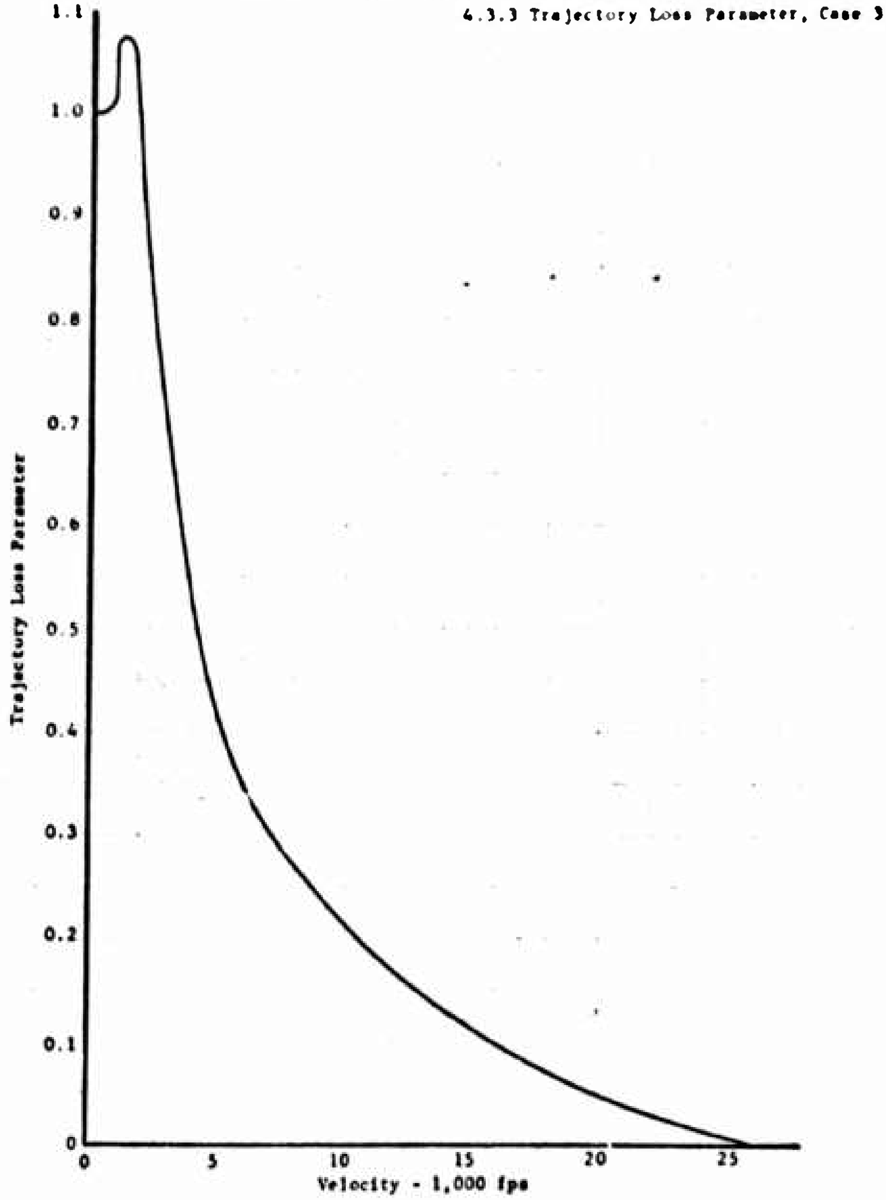
TRAJECTORY LOSS PARAMETER - CASE 3



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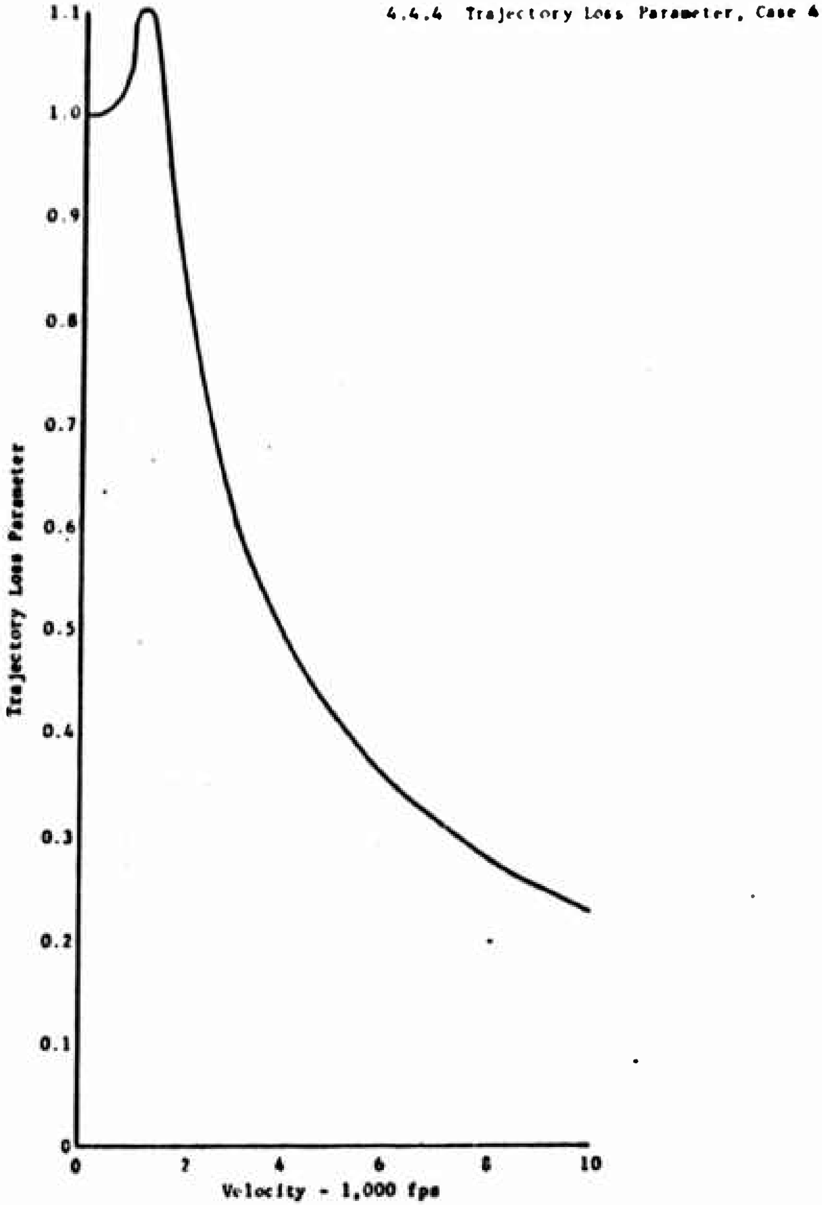
TRAJECTORY LOSS PARAMETER - CASE 4



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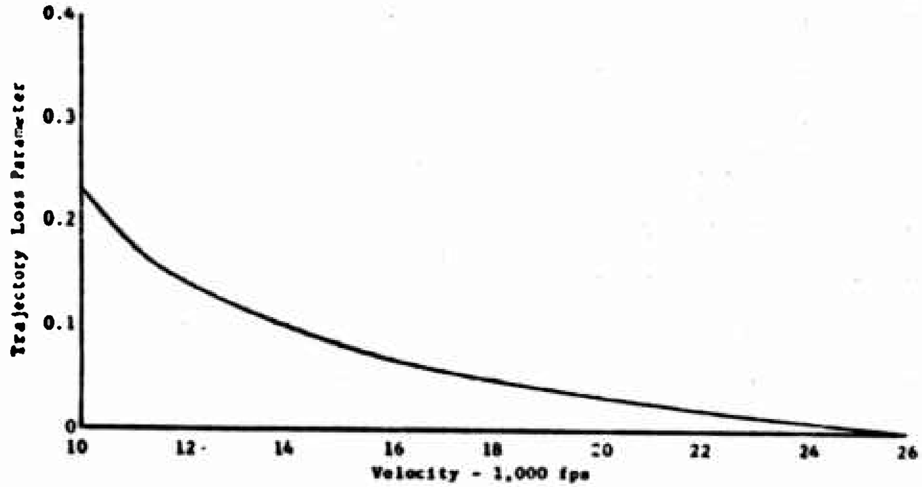
4.4.5 Trajectory Loss Parameter, Cases 5 and 6

TRAJECTORY LOSS PARAMETER, CASES 5 AND 6



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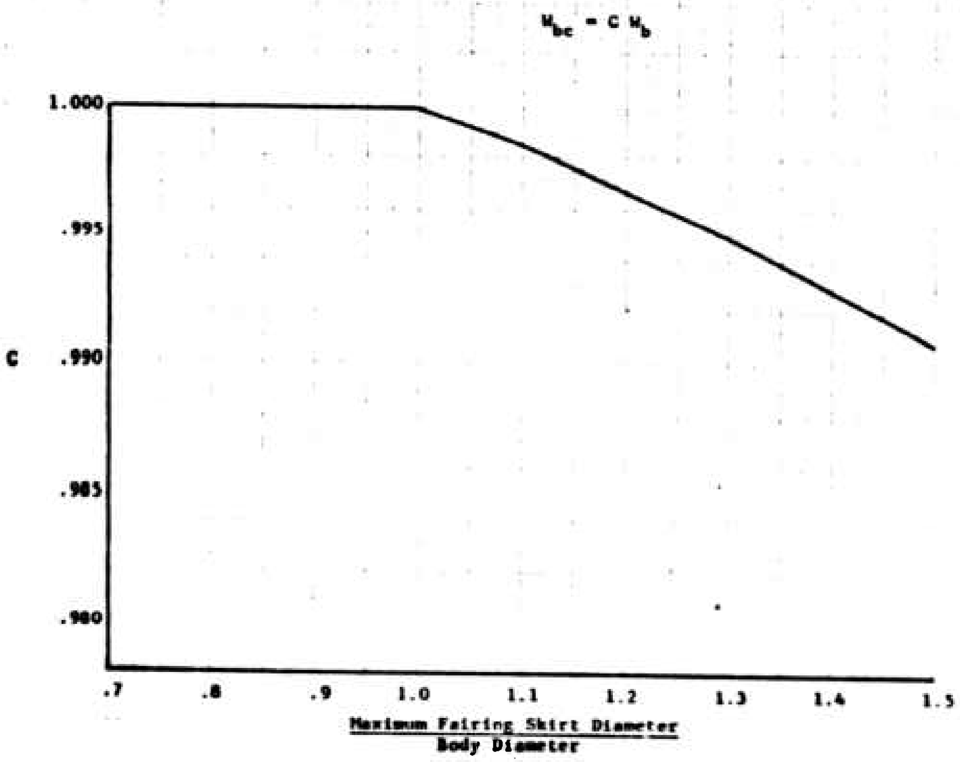
4.5 Burnout Weight Correction Factors*

* No burnout weight correction factor curves are required for Cases #2 and #6; the correction factor, C, shall be assumed to be 1.0

STAGE BURNOUT WEIGHT CORRECTION FACTOR, CASE 1

4.5.1 Burnout Weight Correction Factor, Case 1



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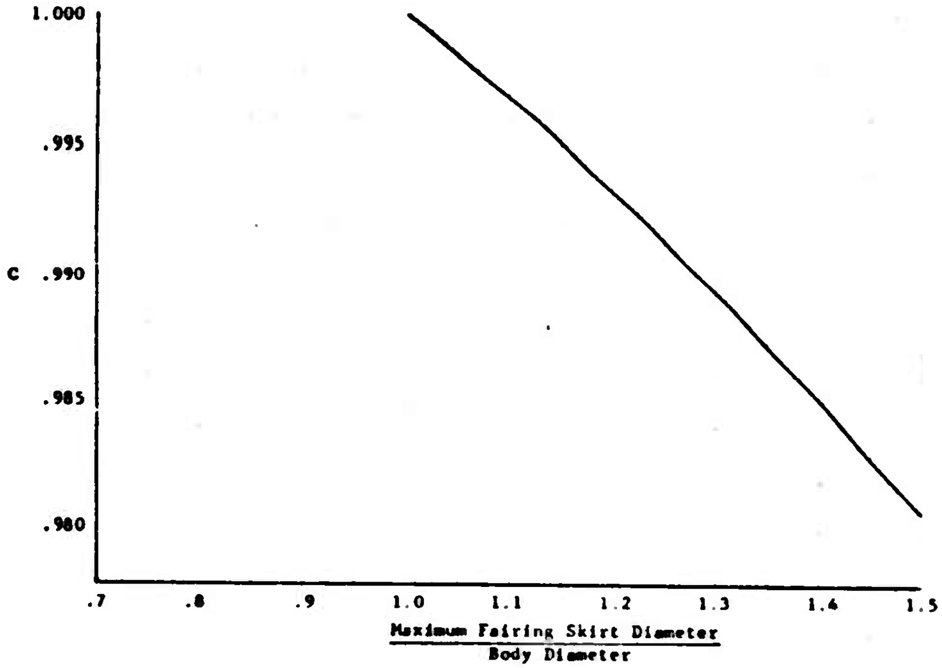
UNCLASSIFIED

STAGE BURNOUT WEIGHT CORRECTION FACTOR, CASE 3

4.5.2 Burnout Weight Correction Factor, Case 3

$W_{bc} = C\, W_b$



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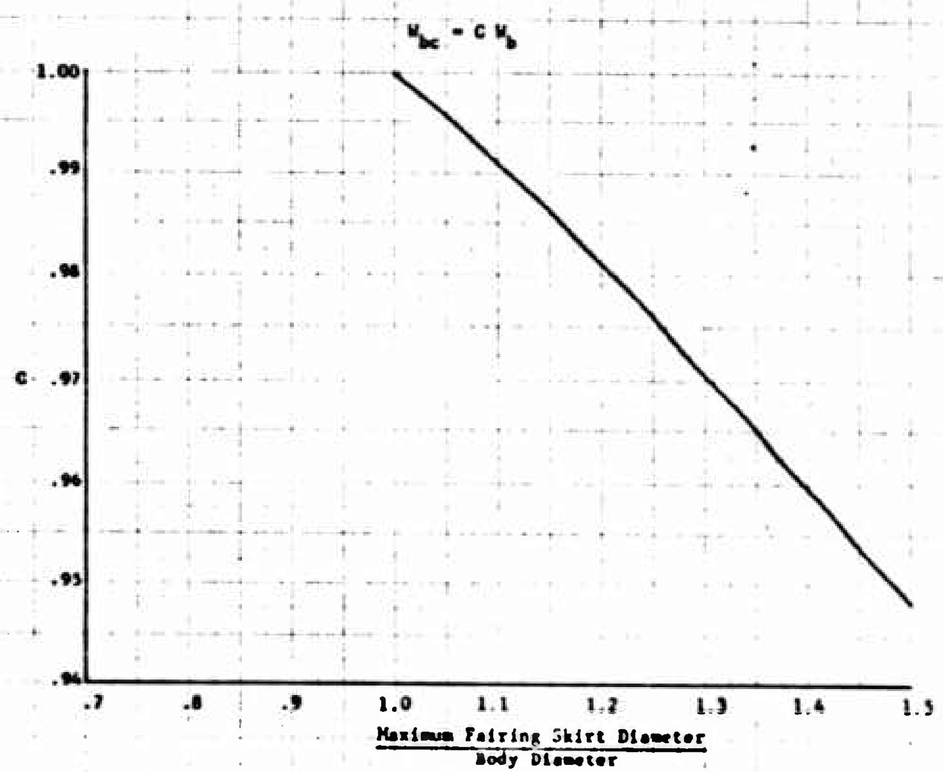
STAGE BURNOUT WEIGHT CORRECTION FACTOR - CASE 4

4.5.4 Burnout Weight Correction Factor, Case 4



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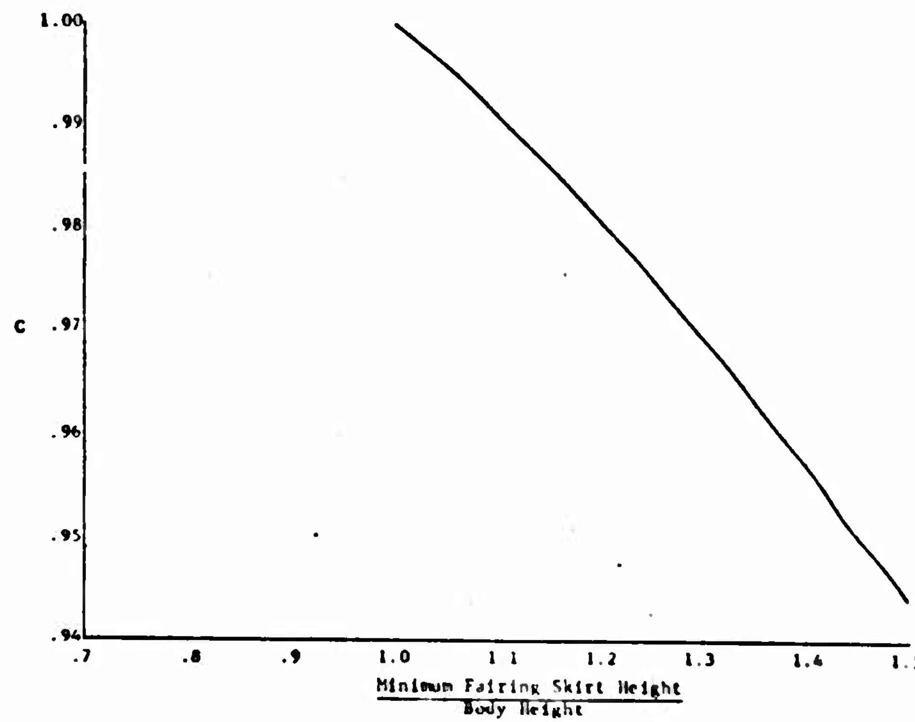
STAGE BURNOUT WEIGHT CORRECTION FACTOR, CASE 5

4.5.3 Burnout Weight Correction Factor, Case 5

$W_{bc} = C\, W_b$



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4.6 Performance Index Equation, and Constants

4.6 CALCULATION OF THE PERFORMANCE INDEX

$$W_x = \left\{\frac{W_g - 2A_i}{W_g}\right\} \left\{W_{bc} - K \left[W_e + 1.5\, A_f + 0.02\,(W_g - W_{bc}) \left(1 + \frac{5}{R}\right) \left(\frac{500{,}000}{W_g - W_{bc}}\right)^{1/3}\right]\right\}$$

where W_x = the Performance Index

W_g = the stage gross weight (lbm)

A_i = the interstage surface area (ft^2)

A_{bc} = the corrected stage burnout weight (lbm)

K = the stage weight-growth constant

W_e = the engine installation weight, including modules, thrust structure, propellant lines, etc (lbm)

A_f = the fairing surface area (ft^2)

R = the oxidizer-to-fuel tank-mixture ratio, by weight

Case:	1	2	3	4	5	6
W_g:	800,000	250,000	800,000	1,280,000	500,000	250,000
K:	1.0	1.0	1.0	2.5	3.0	3.0

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B. VEHICLE APPLICATIONS PACKAGE, 350K MODULE

(U) The specified information used for the 350K application study cases is essentially the same as that presented in the "Vehicle Application Package, 250K Module." The differences in the two packages are in engine thrust level, vehicle gross weight, number of engines per case, and vehicle stage dimensions. The gross weight and number of engines per case (obtained from the Aerospace report "Vehicle Application Package, 350K Module") are presented in table XXX.

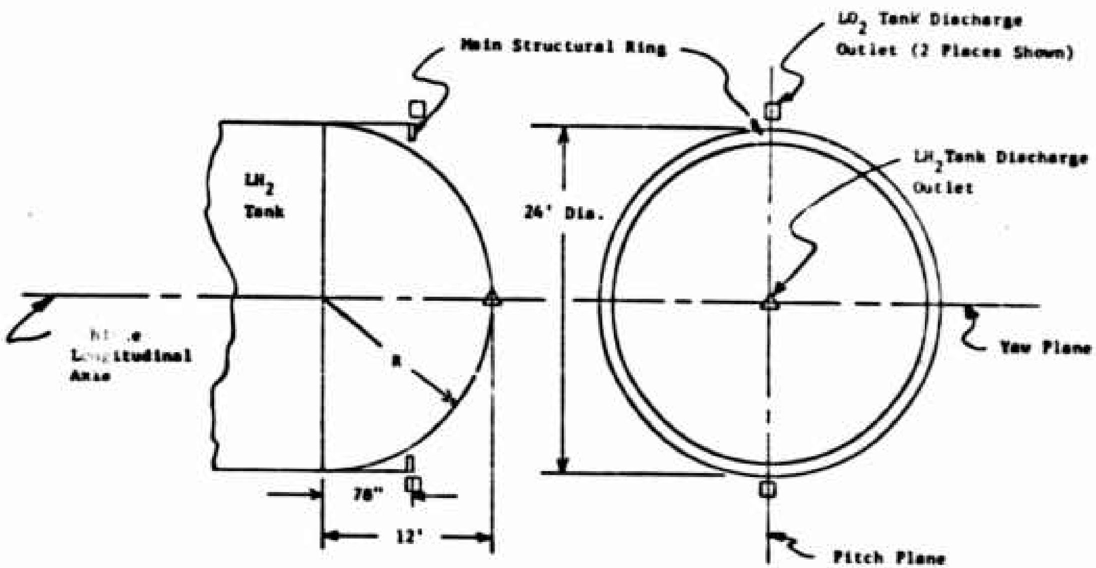
(U) Stage dimensions for the 350K Case such as thrust structure mounting distances, propellant tank size, and feed line location distances are presented in stage installation diagrams for all six cases in figure 156 through 161, which are from the referenced Aerospace report.

(U) Table XXX. Gross Weight and Number of Engines for the Application Study

Case	Gross Weight, lb	Number of Engines
1	900,000	4
2	350,000	1
3	900,000	4
4	1,345,000	6
5	350,000	1
6	350,000	1

CASE I

Gross Takeoff Weight = 900,000 lbs.
Four (4) Engine Modules



Notes:

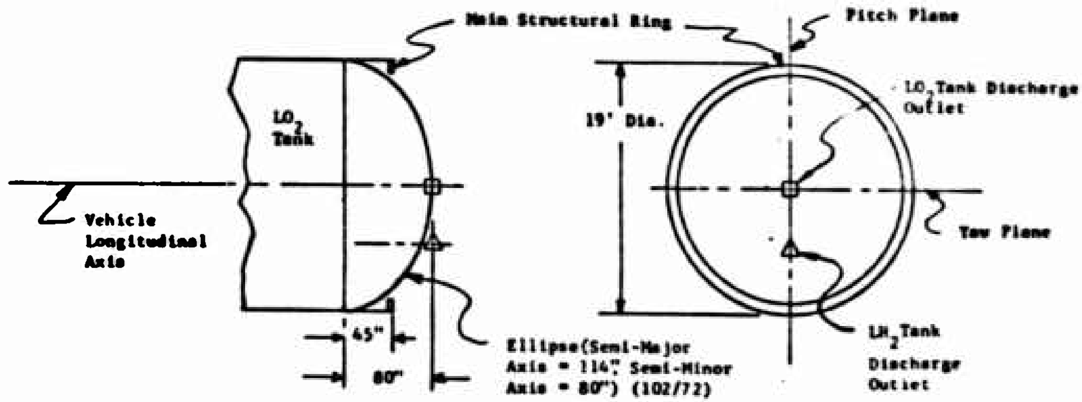
(1) Thrust Mount Structure Shall Be Attached To Main Structural Ring (No. of Points & Location To Be Determined by Contractor)

(2) Up To Four (4) LO_2 Tank Discharge Outlets May Be Utilized By Contractor (Location To Be Symmetrical)

Figure 156. Stage Dimensions - Case 1 (350K Module)

CASE 2

Stage Ignition Weight = 350,000 lbs.
One (1) Upper Stage Engine Module @ Fv = 350,000 lbs.



Notes:

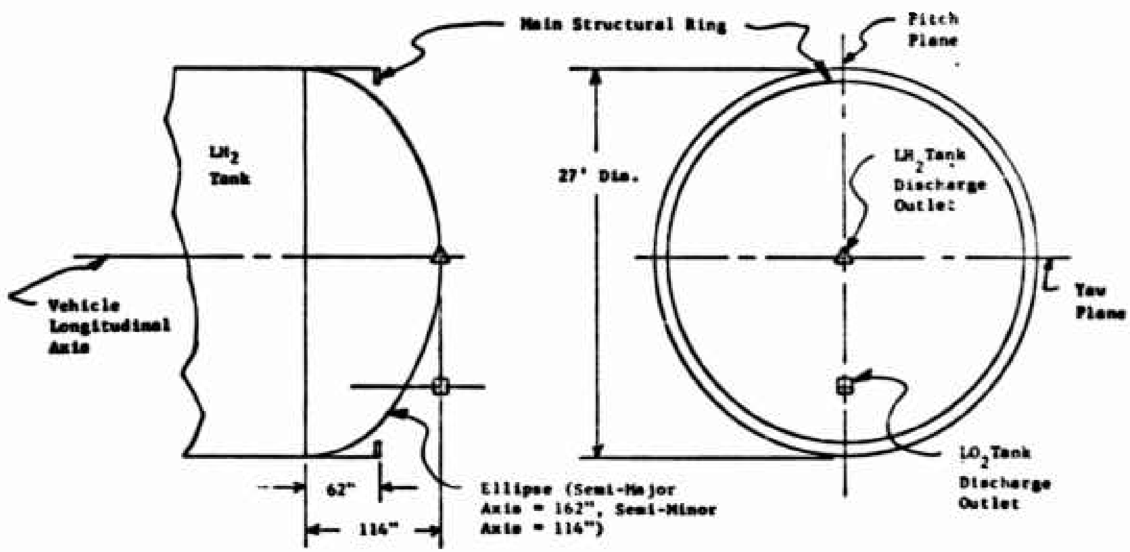
(1) Thrust Mount Structure Shall Be Attached To Main Structural Ring (No. of Points & Location To Be Determined By Contractor)

(2) Radial & Angular Location of LH_2 Tank Discharge Outlet Optional Within Constraints Of Main Structural Ring Diameter.

Figure 157. Stage Dimensions - Case 2 (350K Module)

CASE 3

Gross Takeoff Weight = 900,000 lbs.
Four (4) Engine Modules



Notes:

(1) Thrust Mount Structure Shall Be Attached To Main Structural Ring (No. of Points & Location To Be Determined By Contractor)

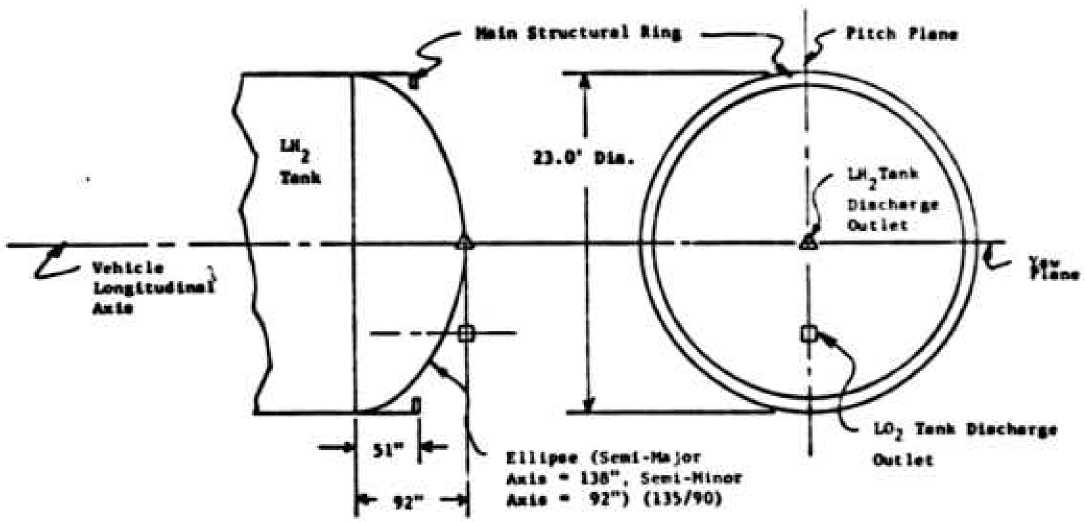
(2) Up To Four(4) LO_2 Tank Discharge Outlets May Be Utilized By Contractor (Locations To Be Symmetrical). Radial Location Optional Within Constraints of Main Structural Ring Diameter.

Figure 158. Stage Dimensions - Case 3 (350K Module)

CASE 4

Gross Takeoff Weight =n

Gross Takeoff Weight = 1,345,000 lbs.
Six (6) Engine Modules



Notes:

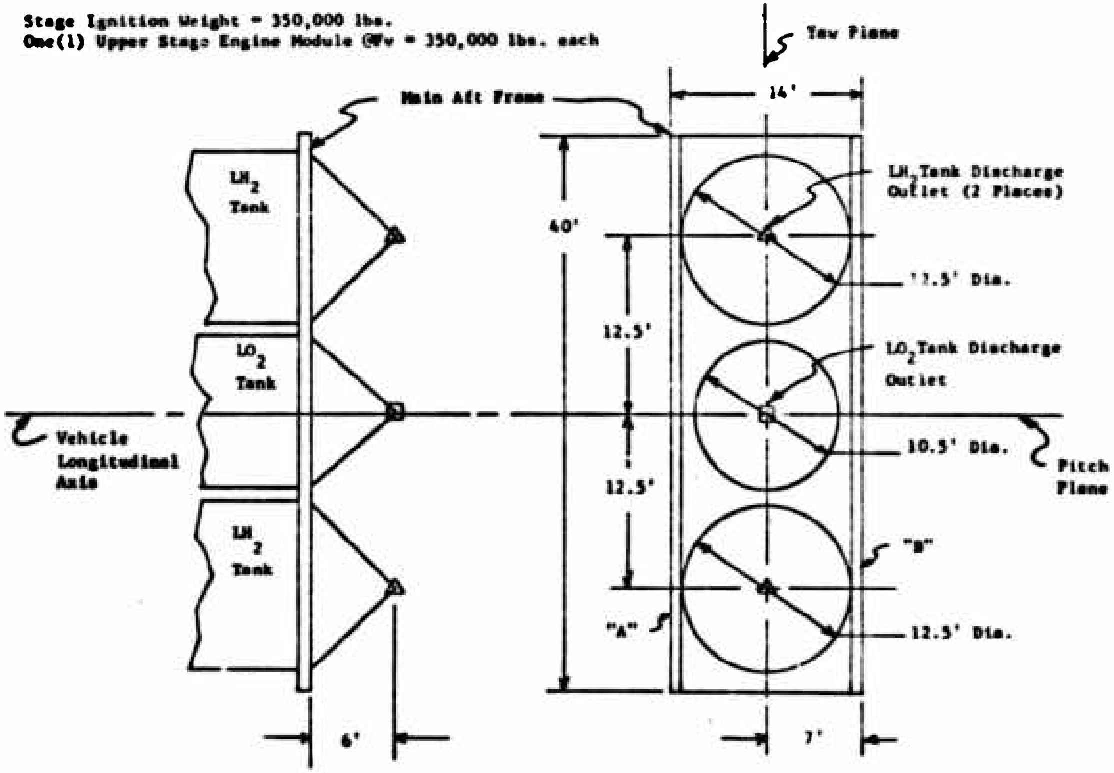
(1) Thrust Mount Structure Shall Be Attached To Main Structural Ring (No. of Points & Location To Be Determined By Contractor)

(2) Up To Three(3) LO_2 Tank Discharge Outlets May Be Utilized By Contractor (Locations To Be Symmetrical). Radial Location Optional Within Constraints Of Main Structural Ring Diameter.

Figure 159. Stage Dimensions - Case 4 (350K Module)

CASE 5

Stage Ignition Weight = 350,000 lbs.
One(1) Upper Stage Engine Module @Fv = 350,000 lbs. each



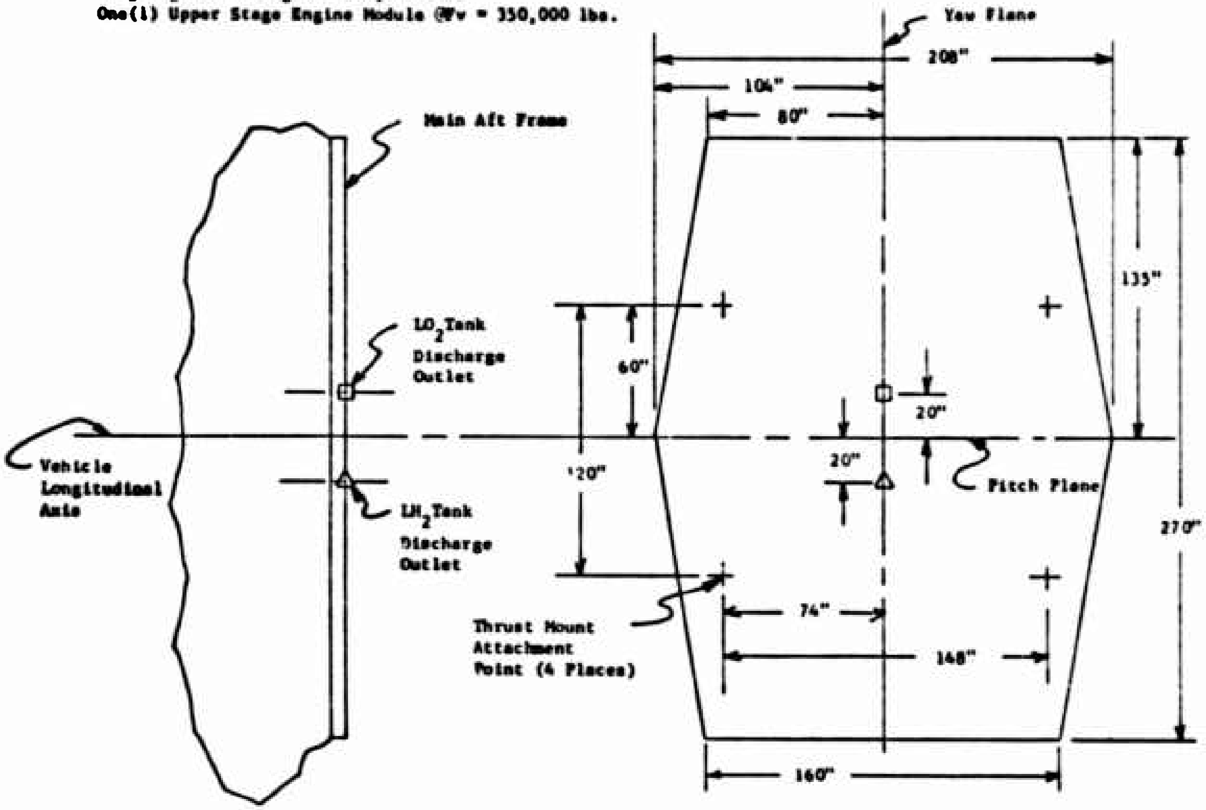
Notes:

(1) Thrust Mount Structure Shall Be Attached To Members "A" & "B" of Main Aft Frame. No. Of Points & Location To Be Determined By Contractor.

Figure 160. Stage Dimensions - Case 5 (350K Module)

CASE 6

Stage Ignition Weight = 350,000 lbs.
One(1) Upper Stage Engine Module @Fv = 350,000 lbs.



Notes:

(1) Thrust Mount Structure Shall Be Attached To Main Aft Frame at Four (4) Points as Indicated.

Figure 161. Stage Dimensions - Case 6 (350K Module)

APPENDIX II
PARAMETRIC ENGINE DATA

(U) This appendix contains parametric performance, size, and weight data used for the application studies. Data are included for the various nozzle concepts and configurations used in the various portions of the study.

A. TWO-POSITION LIGHTWEIGHT NOZZLE

(C) Data presented in this paragraph are for high-pressure two-position, bell nozzle engines, utilizing the pump-fed staged combustion (preburner) cycle. Performance is based on the use of dump cooling downstream of an expansion ratio of 35. For high expansion ratio nozzles, radiation cooling is used aft of the lowest expansion ratio permitted by heat flux. (This expansion ratio varies over the parametric range, but is approximately 200.) Performance, weight, and dimensional data are based on the following:

1. High pressure, staged-combustion, two-position bell nozzle engines
2. Low-speed inducers with:
   a. Minimum hydrogen net positive suction head (NPSH) = 60 ft
   b. Minimum oxygen net positive suction head (NPSH) = 16 ft
3. Continuous throttling capability between 100 and 20% of rated thrust
4. Mixture ratio range of 5 to 7 at all thrust levels
5. Thrust vector control provided by mechanical gimbaling
6. Durability of 10 hours time between overhaul (TBO), 100 reuses, 300 starts, 300 thermal cycles, 10,000 valve cycles
7. Lightweight two-position nozzle which translates to provide high sea level and altitude performance, plus compact packaging.

(U) The parameters included in the parametric data are:

1. Vacuum thrust
2. Chamber pressure
3. Overall engine mixture ratio
4. Overall expansion ratio
5. Nozzle contour
6. Primary expansion ratio.

(U) Values of delivered specific impulse are provided for an altitude range from sea level to vacuum conditions. Engine weight, overall diameter, and length are given. Both the overall length for the nozzle fully extended and the stowed length with the nozzle retracted are presented. The secondary nozzle is the retractable section of the nozzle, and the primary nozzle is the fixed section. The typical engine arrangement and nomenclature are illustrated in figure 162.

(C) Data are presented for engines in which the secondary nozzle skirt is translated from primary expansion ratios of 35, 50, and 80 for a range of overall nozzle area ratios of 50 to 200. In addition, data are presented for primary area ratios that result in the minimum stowed engine length. The primary expansion ratio for minimum stowed length varies with thrust level, chamber pressure, overall expansion ratio, and nozzle contour. For these data, the overall nozzle area ratio range is 80 to 400. The secondary nozzle can be translated over the turbomachinery for those engines that have a primary area ratio greater than 80. For nozzles translating at primary area ratios between 35 and 80, the secondary nozzle is retracted to the point where it is limited by the turbomachinery envelope.

1. Constant Chamber Pressure

(C) The data in this section consist of performance, weight, and size information for the following parameters:

1. Chamber pressure, P_c = 3000 psia
2. Thrust - 100K to 350K
3. Primary area ratio - 35 for maximum performance, base, and minimum surface area contours
4. Primary area ratios - 50 and 80 for maximum performance and base contours
5. Primary area ratios for minimum stowed length for maximum performance, base and minimum surface area contours (vacuum applications only, area ratios 100 to 400).

(C) The data at the specific thrust levels of 250K and 350K are primarily to establish the basic modules; however, weight and dimensional data must also be determined for a constant power package (i.e., constant engine flow rate) for various nozzle area ratios. These data can be obtained from the parametric data by reading the curves at the actual vacuum thrust, for example, the use of a 60:1 area ratio nozzle on the common 250K module (area ratio of 250) will develop a vacuum thrust of 240,700 lb. The weight and size data can be determined by reading the parametric data at 240,700 lb vacuum thrust at an area ratio of 60:1. Small variations in thrust have an insignificant effect on specific impulse.

a. Performance

(U) Delivered vacuum specific impulse (I_{vac}) is presented in figures 163 through 168. These data are relatively insensitive to small thrust level variations, therefore, only 250K and 350K thrust is presented. These data are applicable for the small thrust level variations from 250K (350K) encountered in the application study engine optimization.

(C) Performance from sea level to 200,000 ft is shown as a ratio of delivered specific impulse to vacuum specific impulse plotted as a function of altitude. These data are presented in figures 169 through 186 for primary area ratios of 35, 50, and 80. These data are independent of thrust level for a constant primary area ratio.

(U) The delivered specific impulse at other than vacuum conditions can be calculated using these data. The delivered specific impulse at any altitude up to 200,000 ft (above 200,000 ft is considered vacuum conditions) is calculated by:

$$I_{s_{alt}} = \frac{I_{s_{alt}}}{I_{vac}} I_{vac}$$

where:

$I_{s_{alt}}$ = Delivered specific impulse at the altitude of interest

$I_{s_{alt}}/I_{vac}$ = Ratio of altitude to vacuum performance for the altitude and engine conditions of interest

I_{vac} = Delivered vacuum specific impulse for the engine conditions and secondary area ratio of interest

also:

$$F_{alt} = \frac{I_{s_{alt}}}{I_{vac}} F_{vac}$$

In instances where I_s/I_{vac} is plotted versus pressure ratio, the pressure ratio is calculated by:

$$PR = \frac{P_c}{P_a}$$

where:

P_c = Chamber pressure, psia

P_a = Ambient pressure at the altitude of interest, psia.

(U) The I_s/I_{vac} curves present altitude performance for the engines with the secondary nozzle extended at high altitudes and with the secondary nozzle retracted at low altitudes. Low altitude primary nozzle data have not been included for instances where the primary nozzle exhaust would be expected to reattach to the secondary shroud.

b. Weight

(U) Engine weight is presented in figures 187 through 196.

c. Dimensional Data

(C) Stowed engine length (secondary nozzle retracted) is shown in figures 197 through 206. Overall engine length (secondary nozzle extended) is shown in figures 207 through 209. Secondary nozzle exit diameter is shown in figure 210. This is the engine envelope diameter for engines with an overall expansion ratio equal to 80 or larger. At lower area ratios, the envelope diameter is determined by the diameter of the turbomachinery power package and is approximately equal to 65 in. and 80 in. for the 250K and 350K engines, respectively.

2. Effects of Variable Chamber Pressure

(C) The data used in the evaluation of the design chamber pressure were based on the following:

1. Chamber pressure, P_c = 2000 to 3500 psia
2. Thrust = 200K and 250K
3. Nozzle contour - minimum surface area
4. Primary area ratio = 35 (overall expansion ratio)
5. Primary area ratio for minimum stowed length (overall expansion ratio 80 to 400, vacuum applications only).

Data are included for a lower thrust level so that weight and size data may be obtained for the common module.

a. Performance

(C) Delivered vacuum specific impulse is shown as a function of chamber pressure in figures 211 through 213 for 250K. The ratio of delivered specific impulse at any altitude from sea level to 200,000 ft to delivered vacuum specific impulse is shown as a function of pressure ratio in figures 214 through 216. The performance data given are based on full regenerative cooling of the thrust chamber and nozzle at the 2000 psia chamber pressure point.

b. Weight

(U) Total engine weight is presented as a function of chamber pressure in figures 217 and 218 for 250K thrust. Engine weight data are provided in figures 219 and 220 for 200K thrust. The 200K and 250K thrust data may be interpolated to obtain the weight and dimensional size for required intermediate values of thrust.

c. Dimensional Data

(U) Stowed length is presented in figures 221 and 222, overall length in figure 223, and nozzle exit diameter in figure 224 for 250K thrust. Stowed length is presented in figures 225 and 226, overall length in figure 227, and nozzle exit diameter in figure 228 for 200K thrust.

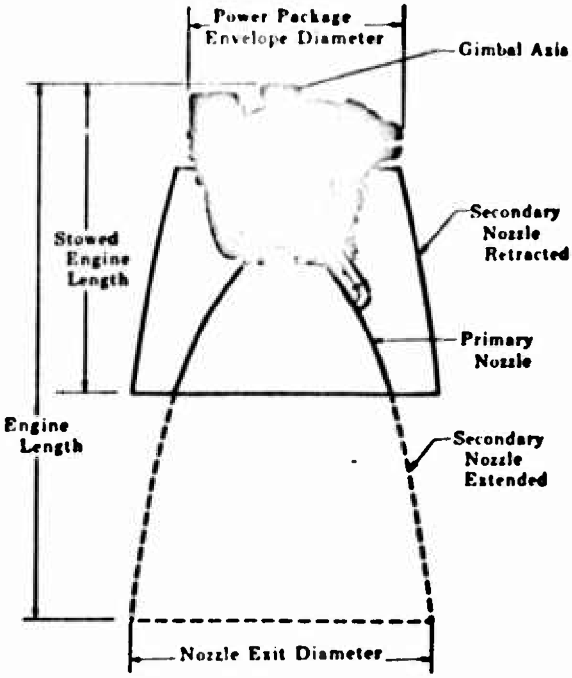


Figure 162. Engine Configuration With Two-Position Nozzle

FD 21152

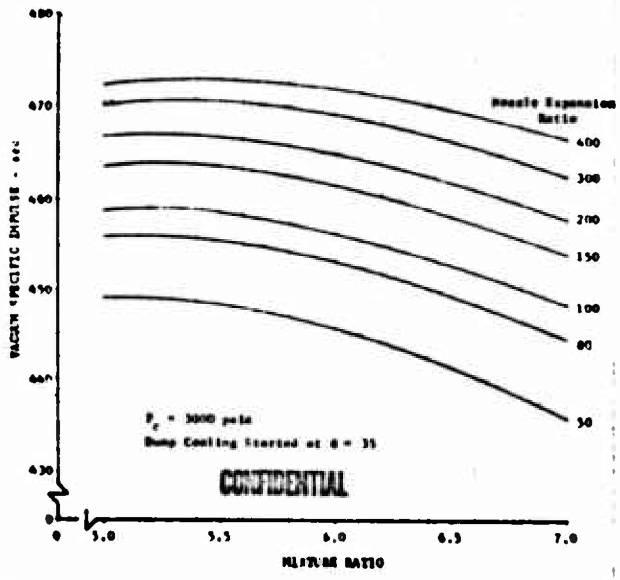


Figure 163. Vacuum Specific Impulse vs Mixture Ratio, Base Contour Two-Position Nozzle (250K Module) DF 57240

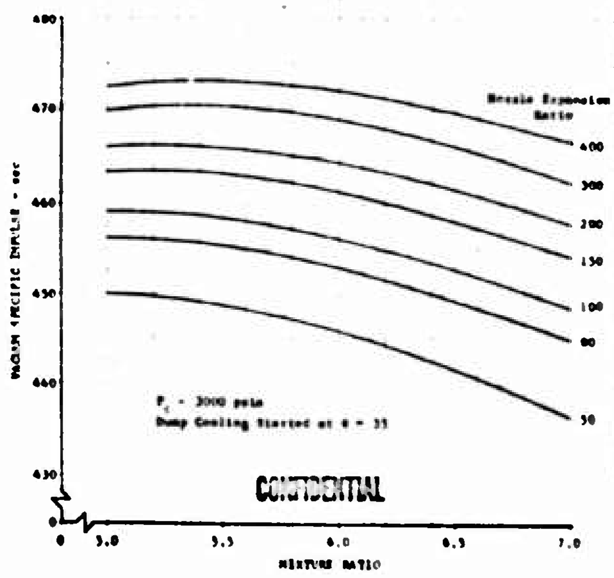


Figure 164. Vacuum Specific Impulse vs Mixture Ratio, Maximum Performance Contour Two-Position Nozzle (250K Module) DF 57241

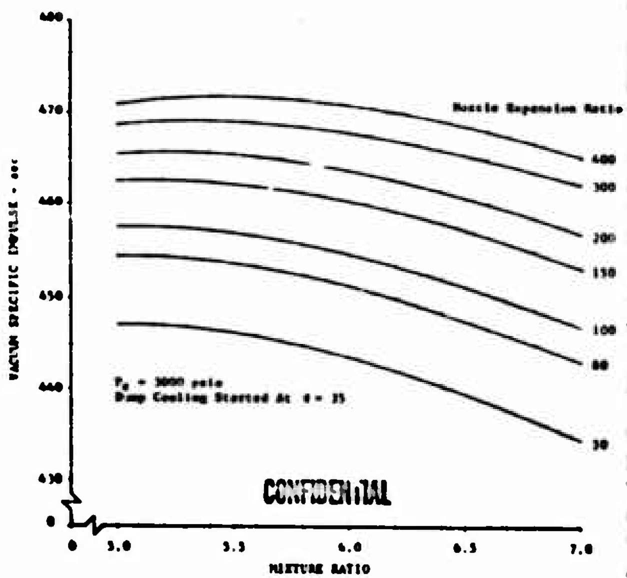


Figure 165. Vacuum Specific Impulse vs Mixture Ratio, Minimum Surface Area Contour Two-Position Nozzle (250K Module) DF 57242

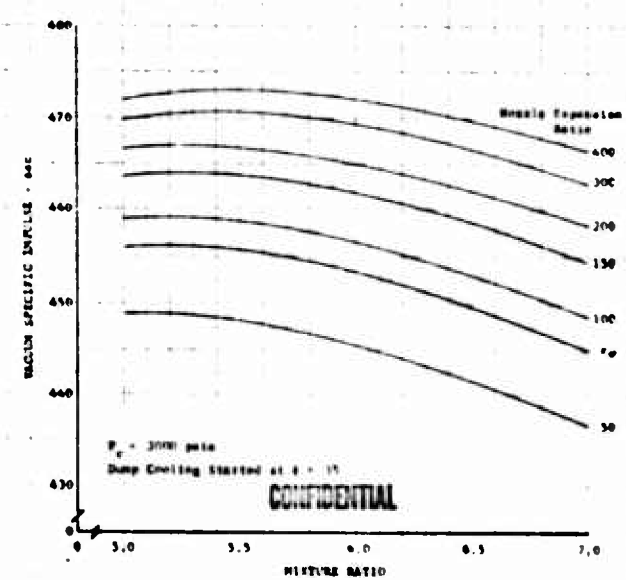


Figure 166. Vacuum Specific Impulse vs Mixture Ratio, Base Nozzle (350K Module) DF 57243

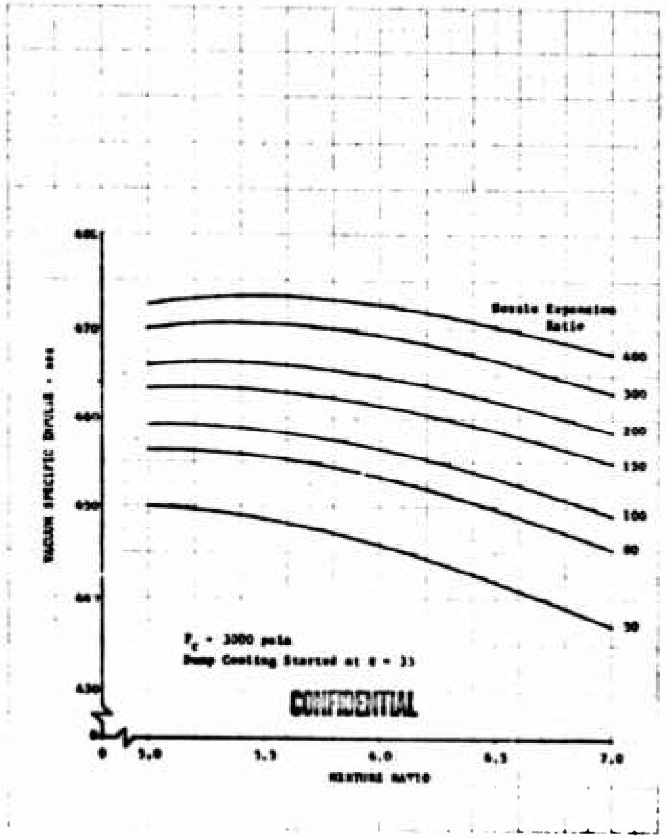


Figure 167. Vacuum Specific Impulse vs Mixture Ratio, Maximum Performance Nozzle (350K Module) DF 57244

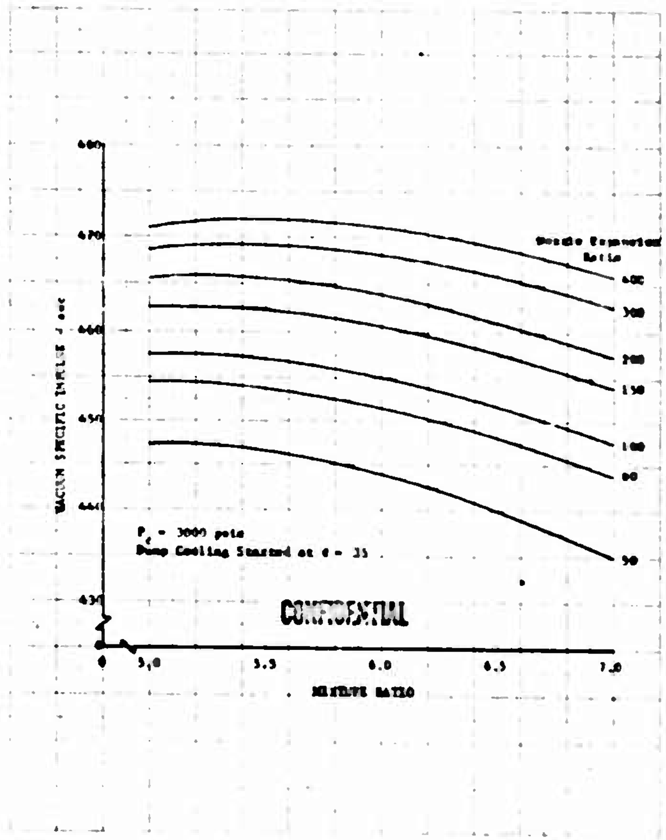


Figure 168. Vacuum Specific Impulse vs Mixture Ratio, Minimum Surface Area Nozzle (350K Module) DF 57245

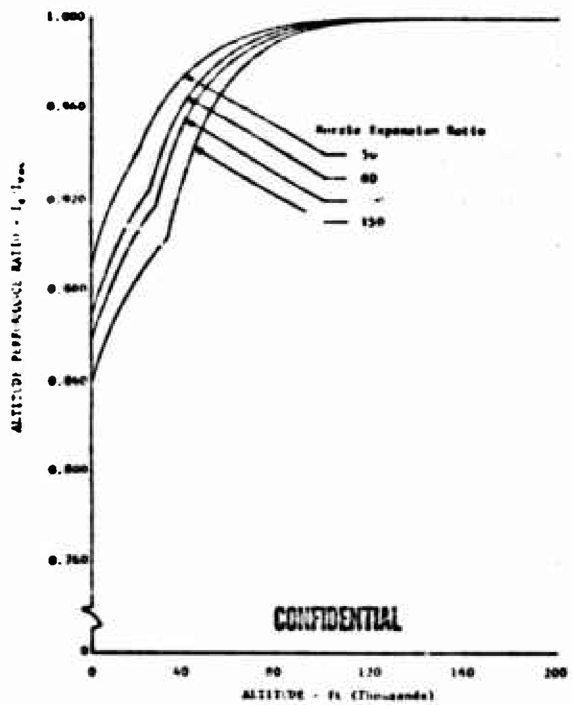


Figure 169. Altitude Performance With Maximum Performance Contour Two-Position Nozzle, Mixture Ratio of 5 ($\epsilon_p = 35$) DF 57252

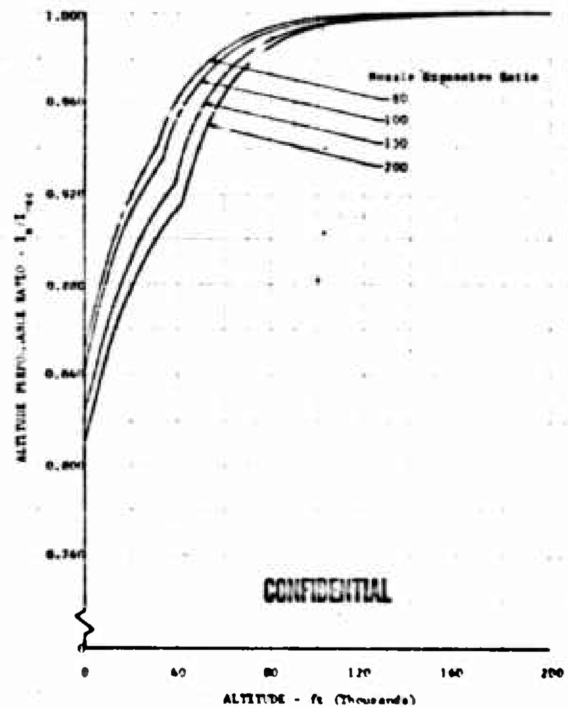


Figure 170. Altitude Performance With Maximum Performance Contour Two-Position Nozzle, Mixture Ratio of 5 ($\epsilon_p = 50$) DF 57248

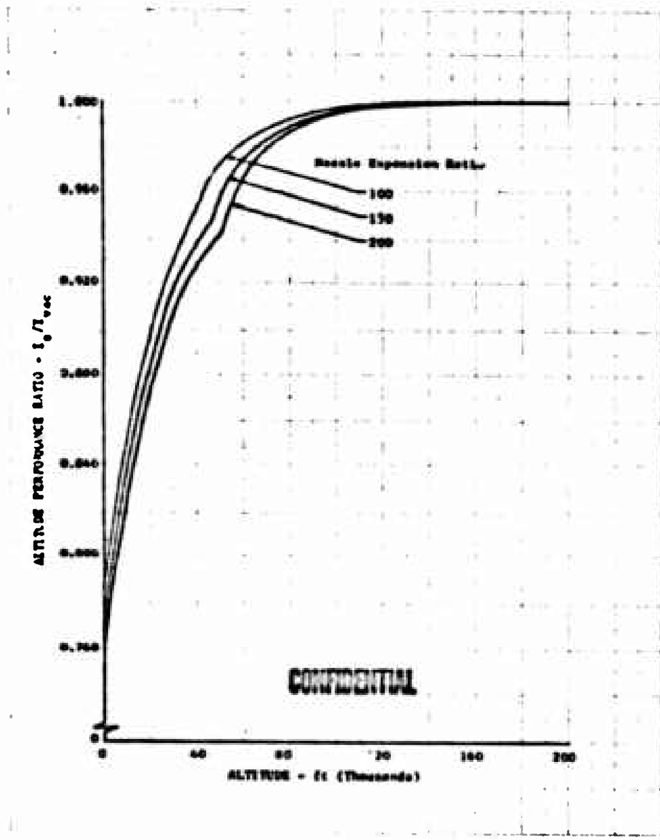


Figure 171. Altitude Performance With Maximum Perfor ance Contour Two-Position Nozzle, Mixture Ratio of 5 (ϵ_p = 80) DF 57249

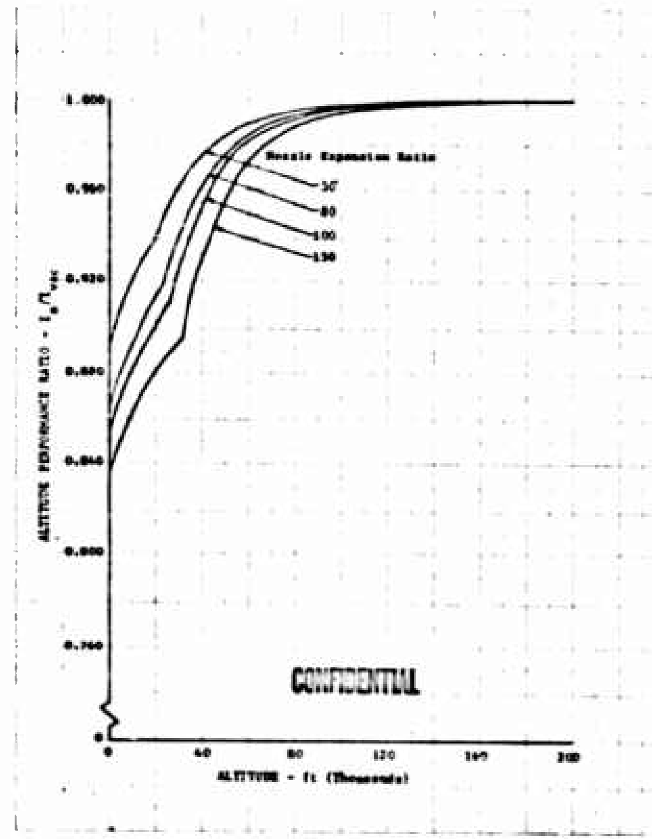


Figure 172. Altitude Performance With Maximum Performance Contour Two-Position Nozzle, Mixture Ratio of 6 (ϵ_p = 35) DF 57250

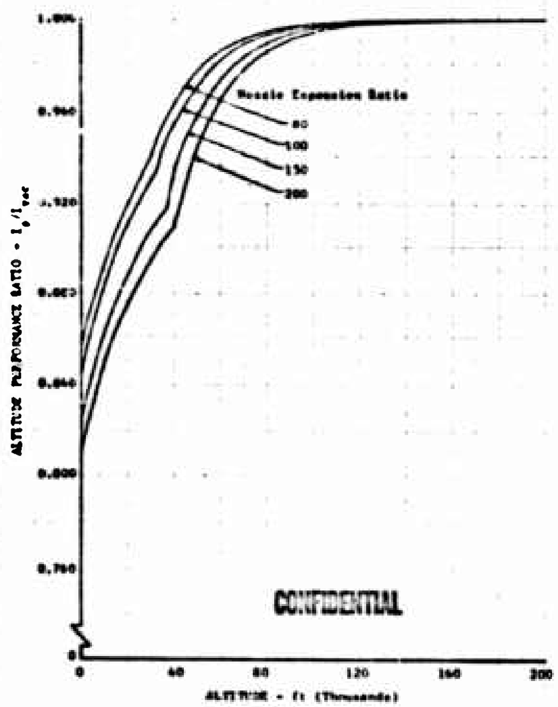


Figure 173. Altitude Performance With Maximum Performance Contour Two-Position Nozzle, Mixture Ratio of 6 ($\epsilon_p = 50$) DF 57251

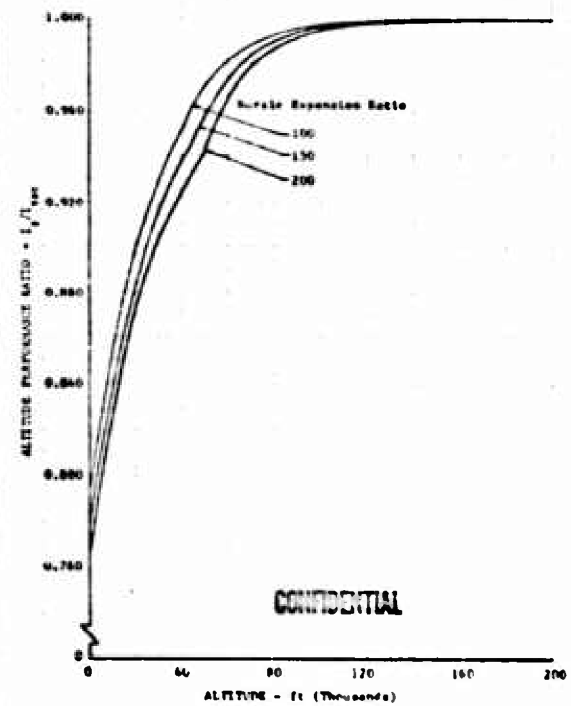


Figure 174. Altitude Performance With Maximum Performance Contour Two-Position Nozzle, Mixture Ratio of 6 ($\epsilon_p = 80$) DF 57253

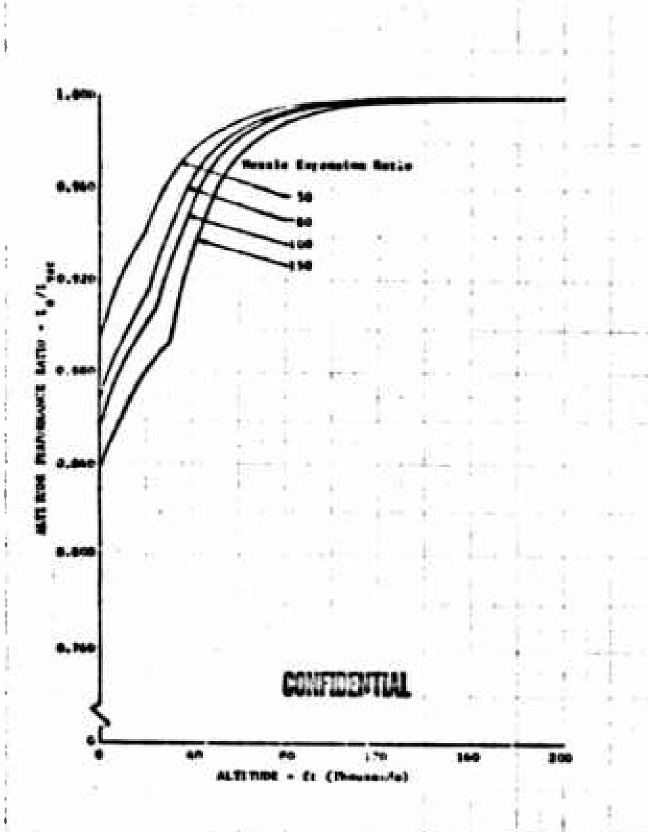


Figure 175. Altitude Performance With Maximum Performance Contour Two-Position Nozzle, Mixture Ratio of 7 (ϵ_p = 35) DF 57254

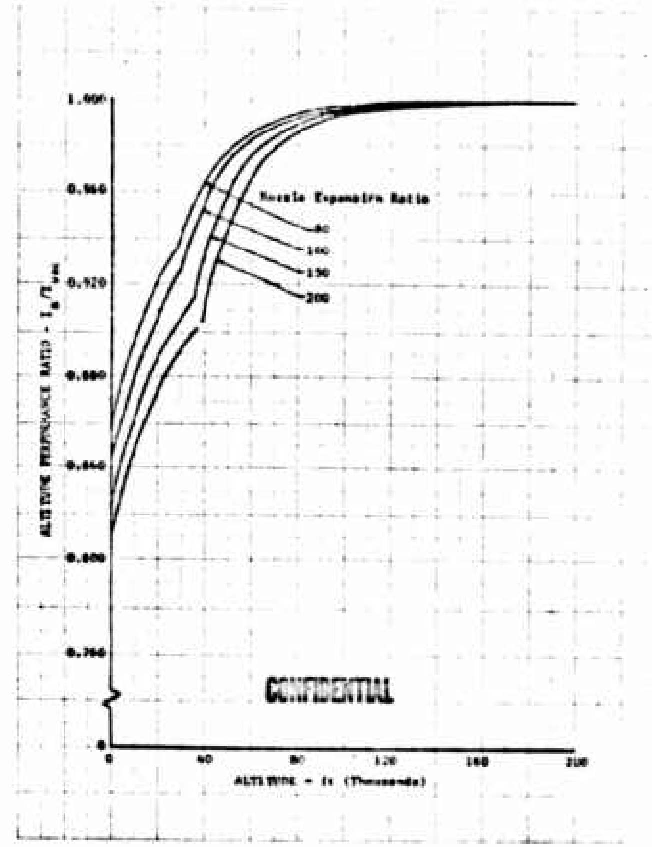


Figure 176. Altitude Performance With Maximum Performance Contour Two-Position Nozzle, Mixture Ratio of 7 (ϵ_p = 50) DF 57255

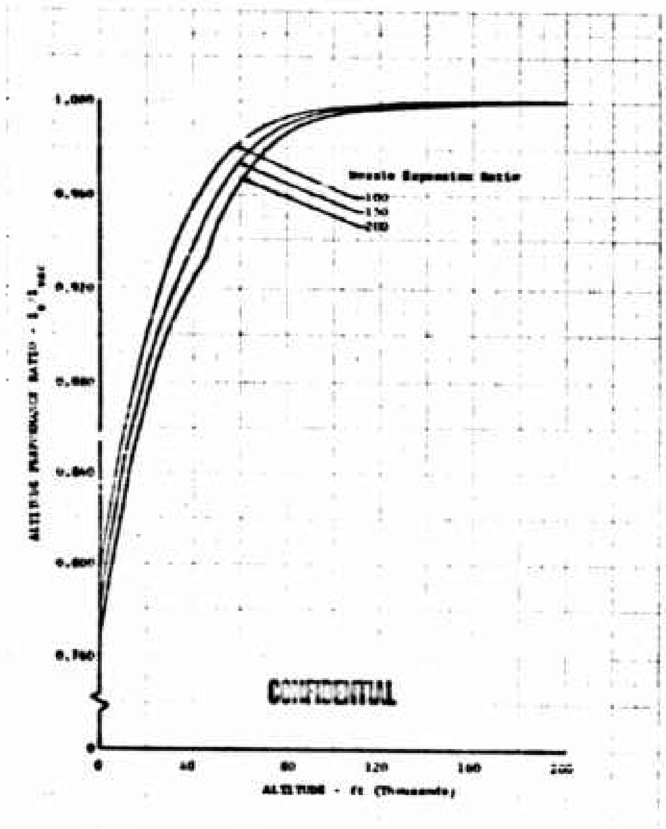


Figure 177. Altitude Performance With Maximum Performance Contour Two-Position Nozzle, Mixture Ratio of 7 (ϵ_p = 80) DF 57265

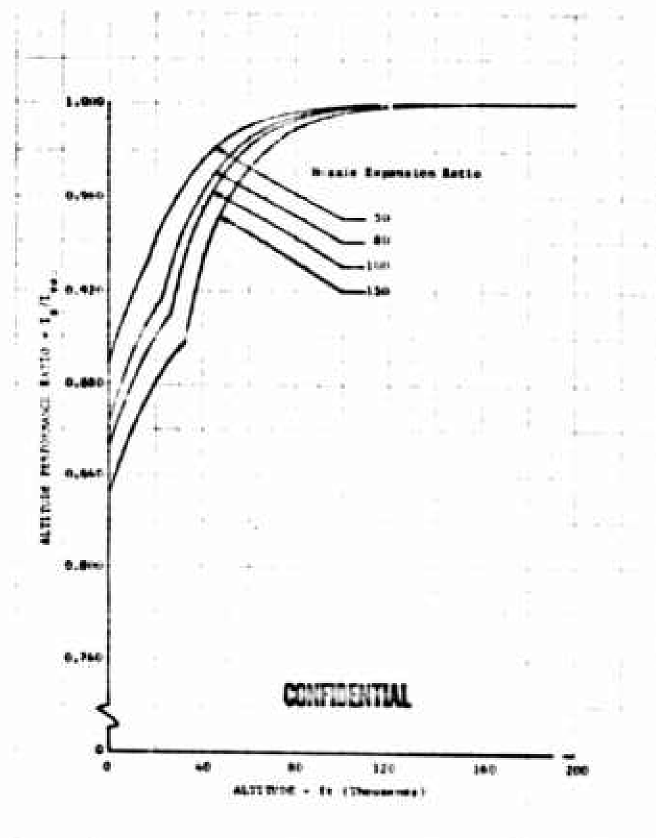


Figure 178. Altitude Performance With Base Contour Two-Position Nozzle, Mixture Ratio of 5 (ϵ_p = 35) DF 57266

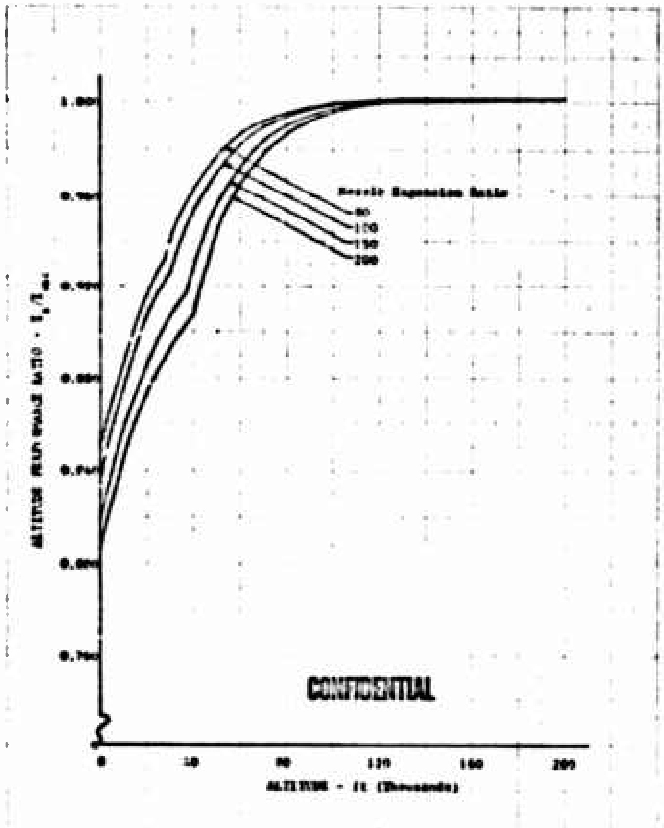


Figure 179. Altitude Performance With Base Contour Two-Position Nozzle, Mixture Ratio of 5 (ϵ_p = 50) DF 57267

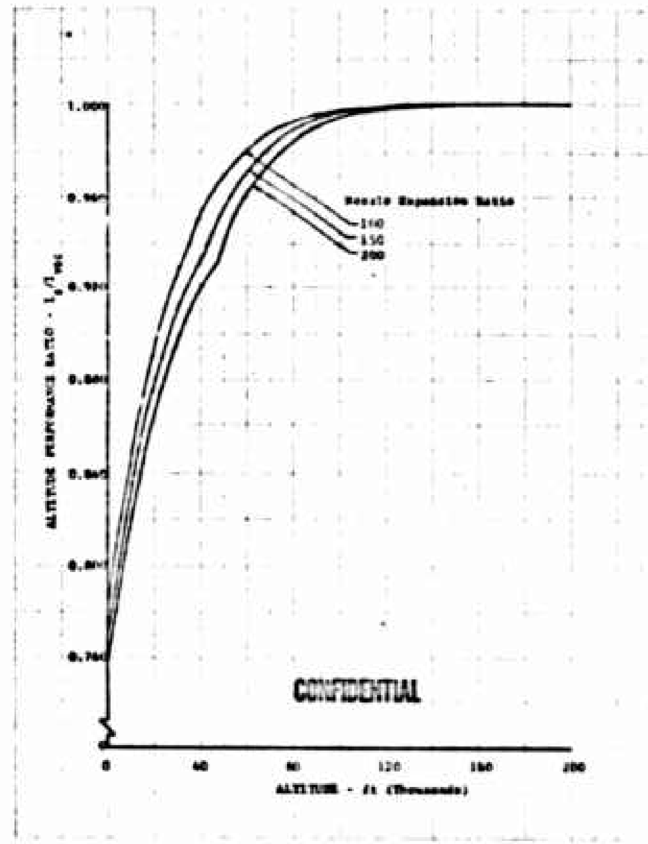


Figure 180. Altitude Performance With Base Contour Two-Position Nozzle, Mixture Ratio of 5 (ϵ_p = 80) DF 57268

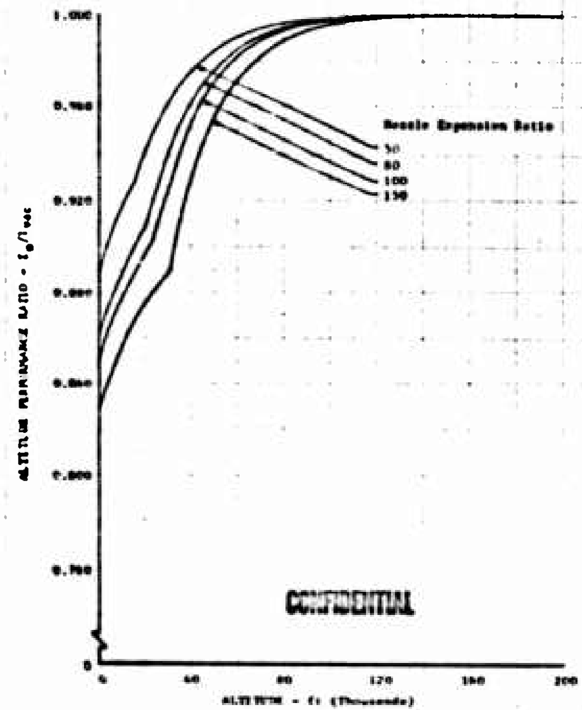


Figure 181. Altitude Performance With Base Contour Two-Position Nozzle, Mixture Ratio of 6 (ϵ_p = 35) DF 57269

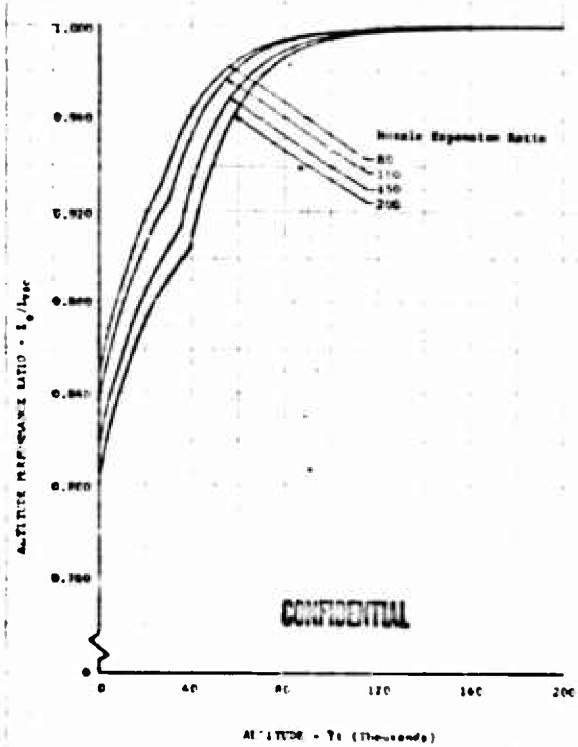


Figure 182. Altitude Performance With Base Contour Two-Position Nozzle, Mixture Ratio of 6 (ϵ_p = 50) DF 57270

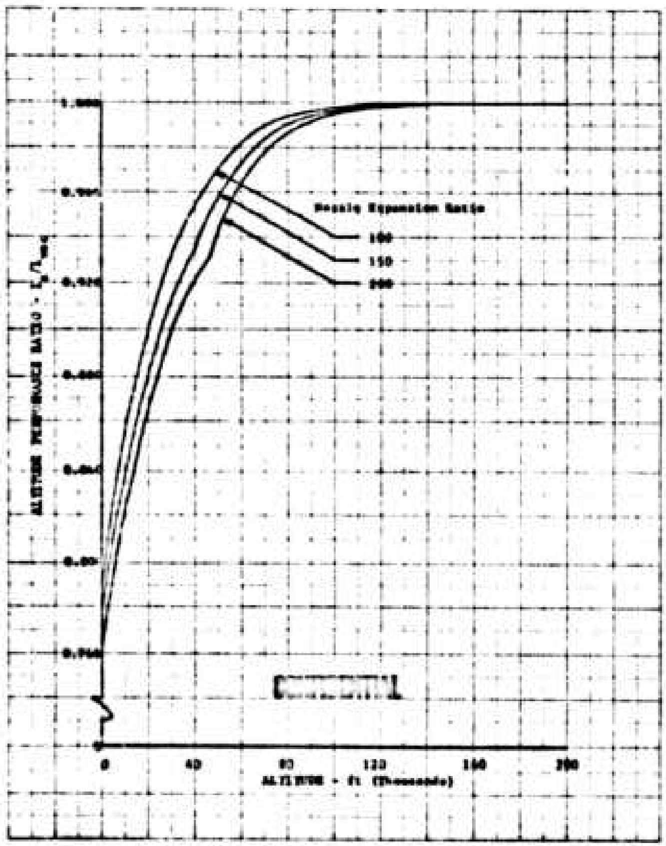


Figure 183. Altitude Performance With Base Contour Two-Position Nozzle, Mixture Ratio of 6 (ϵ_p = 80) DF 57271

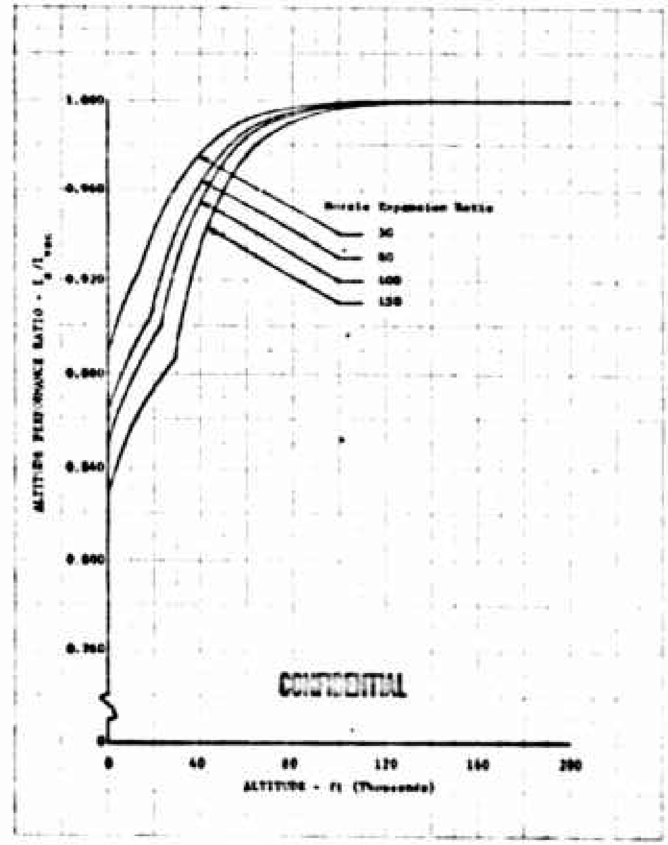


Figure 184. Altitude Performance With Base Contour Two-Position Nozzle, Mixture Ratio of 7 (ϵ_p = 35) DF 5727?

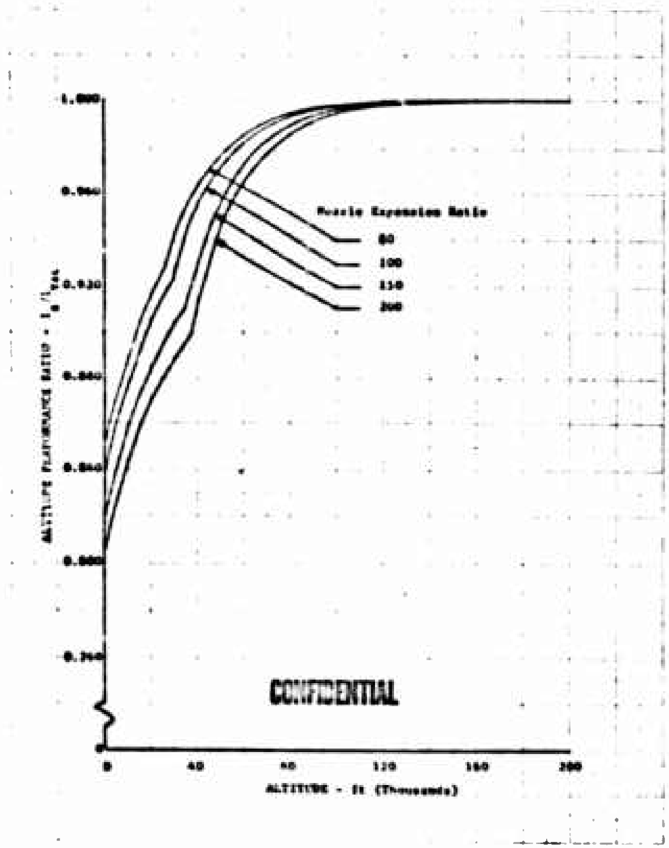


Figure 185. Altitude Performance With Base Contour Two-Position Nozzle, Mixture Ratio of 7 (ϵ_p = 50) DF 57273

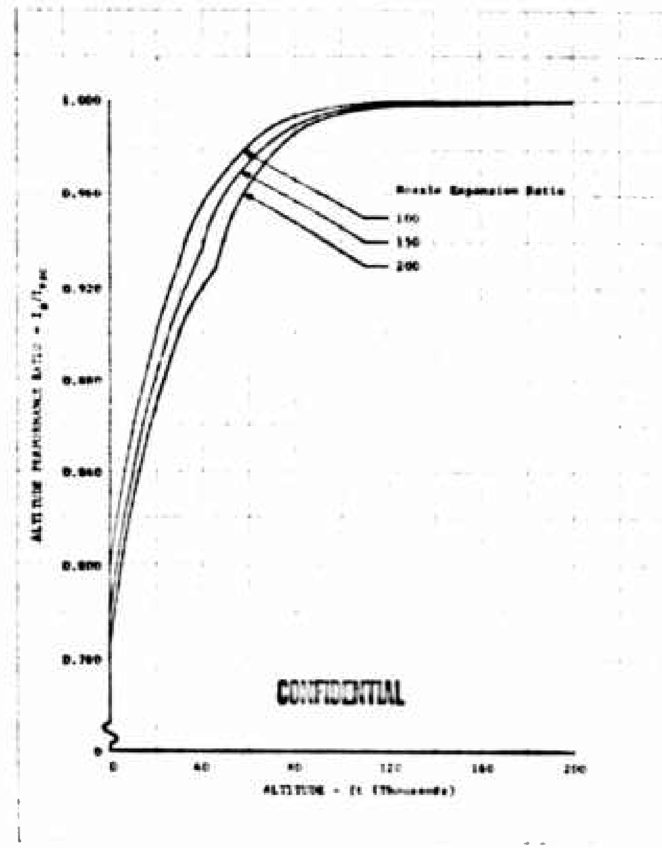


Figure 186. Altitude Performance With Base Contour Two-Position Nozzle, Mixture Ratio of 7 (ϵ_p = 80) DF 57274

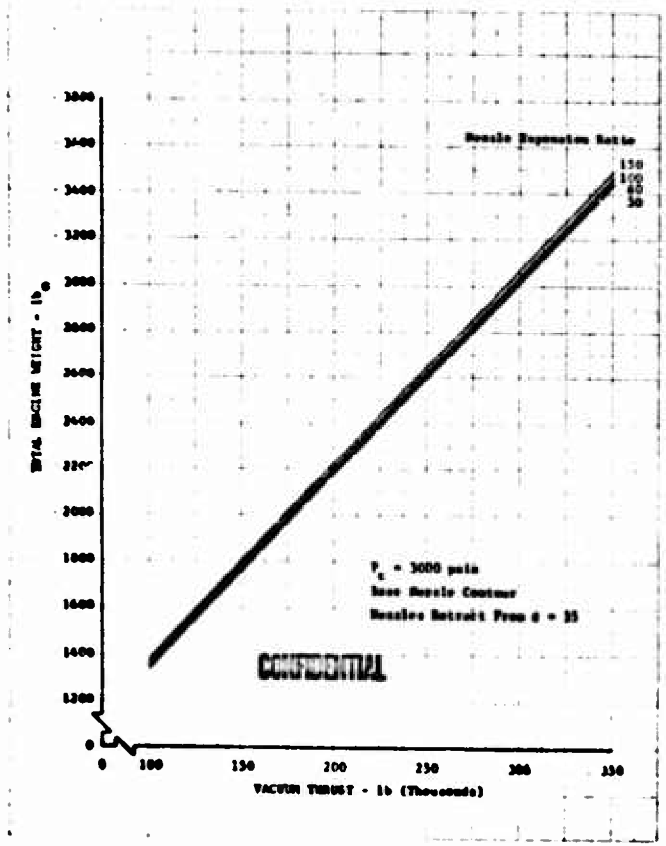


Figure 187. Engine Weight With Base Contour Two-Position Nozzle (ϵ_p = 35) DF 56357

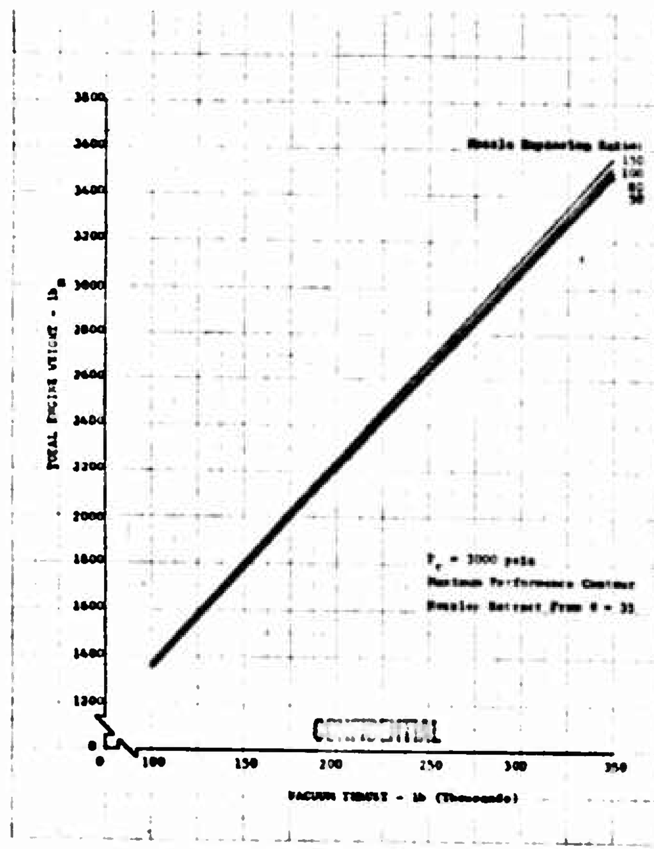


Figure 188. Engine Weight With Maximum Performance Contour Two-Position Nozzle (ϵ_p = 35) DF 56356

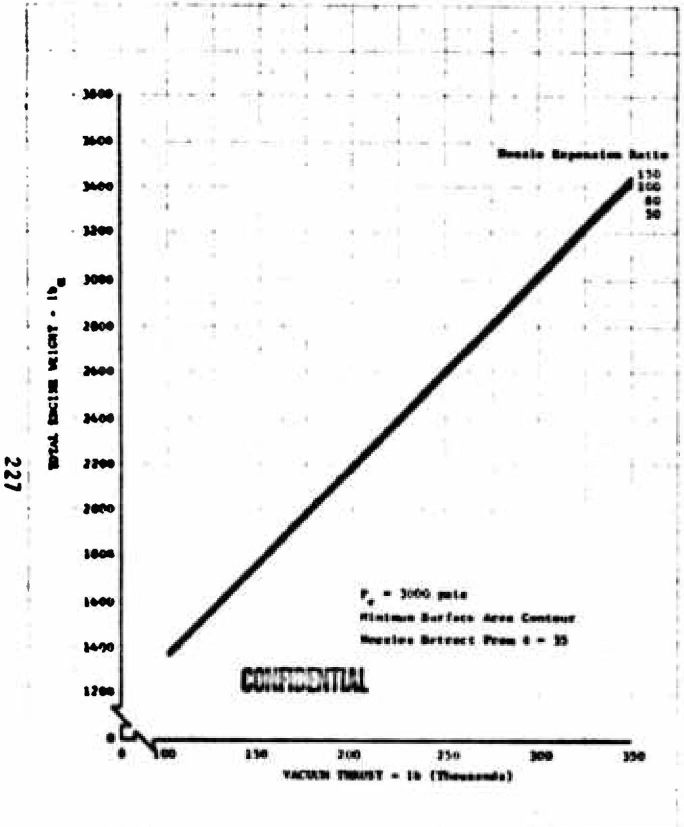


Figure 189. Engine Weight With Minimum Surface Area Contour Two-Position Nozzle (ϵ_p = 35) DF 56358

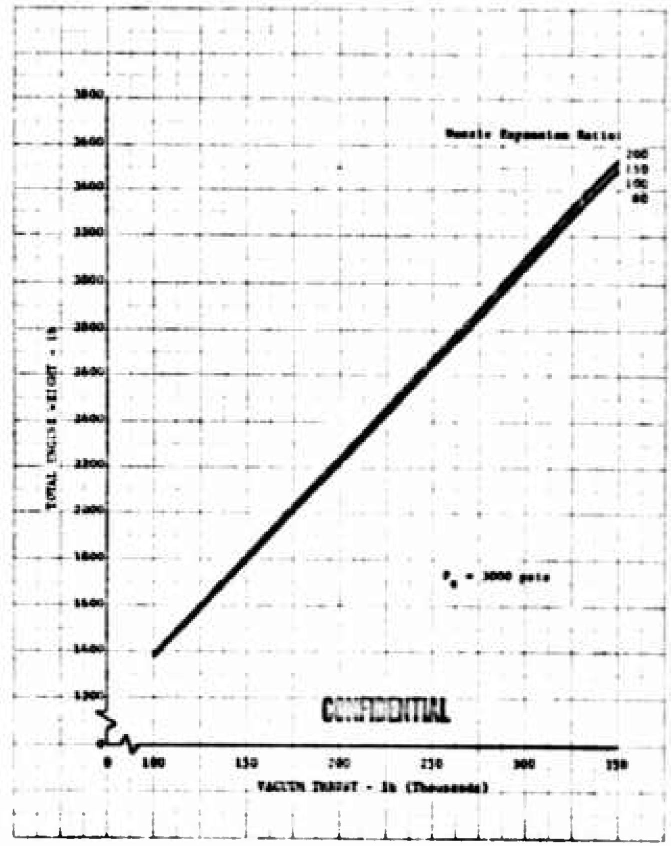


Figure 190. Engine Weight With Base Contour Two-Position Nozzle (ϵ_p = 50) DF 57279

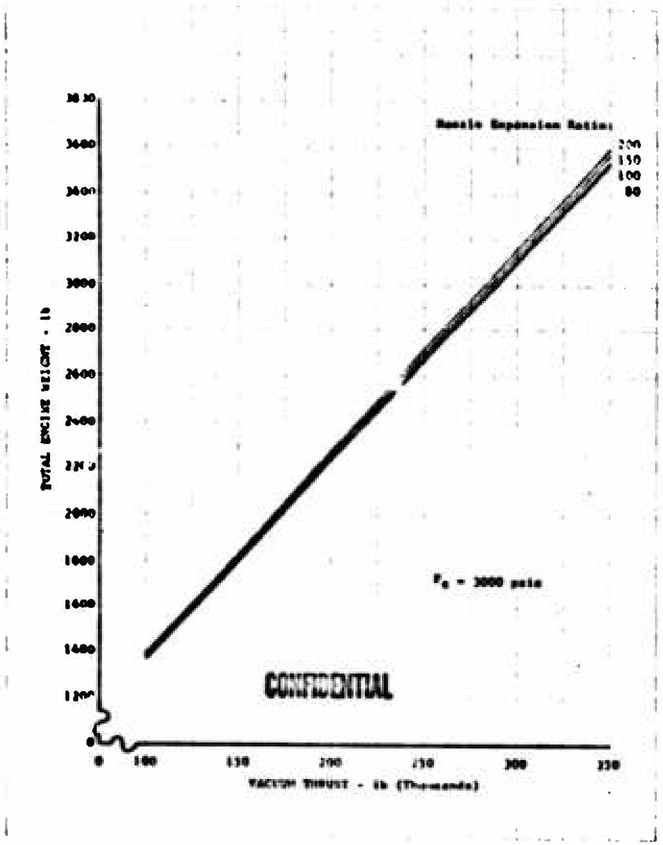


Figure 191. Engine Weight With Maximum Performance Contour Two-Position Nozzle (ϵ_p = 50) DF 572P1

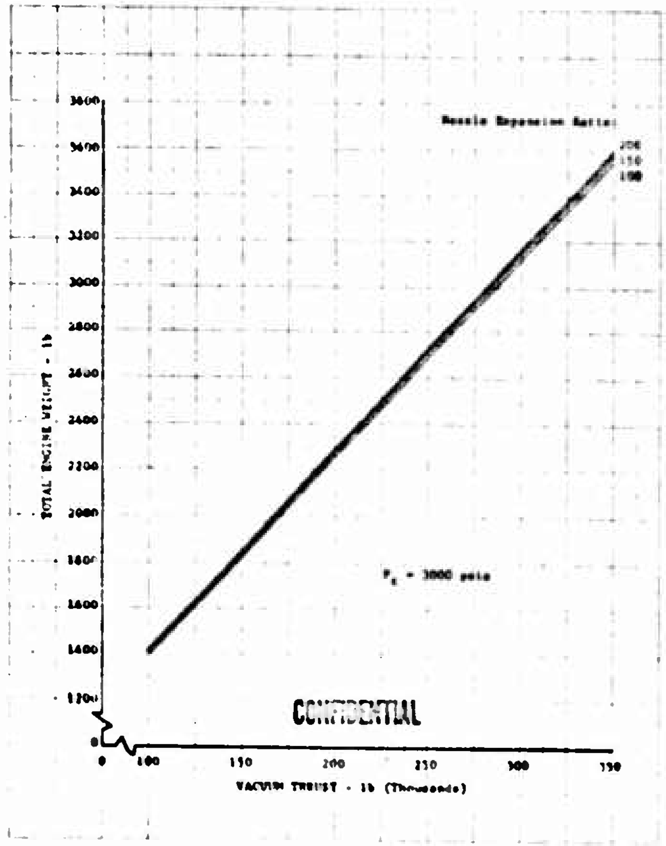


Figure 192. Engine Weight With Base Contour Two-Position Nozzle (ϵ_p = 80) DF 57280

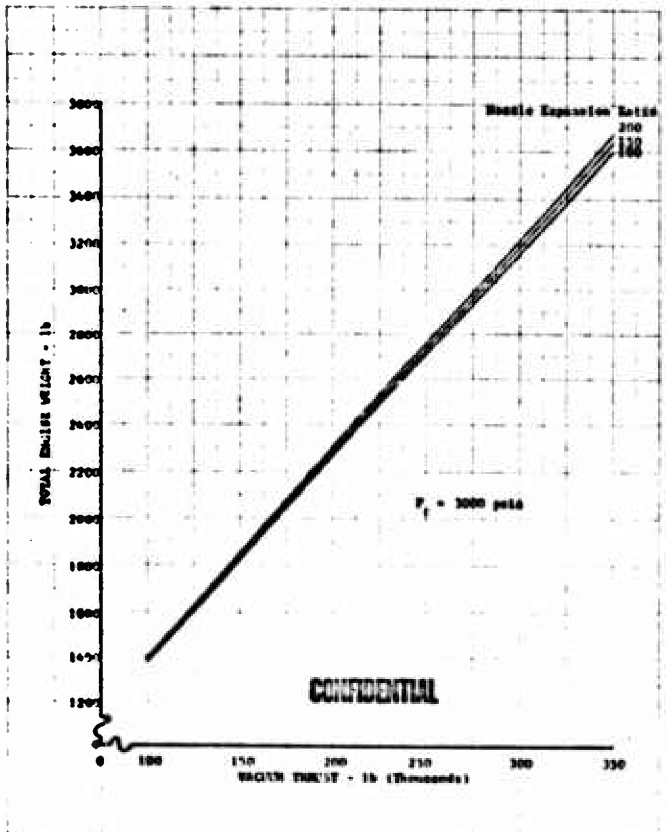


Figure 193. Engine Weight With Maximum Performance Contour Two-Position Nozzle (ϵ_p = 80) DF 57282

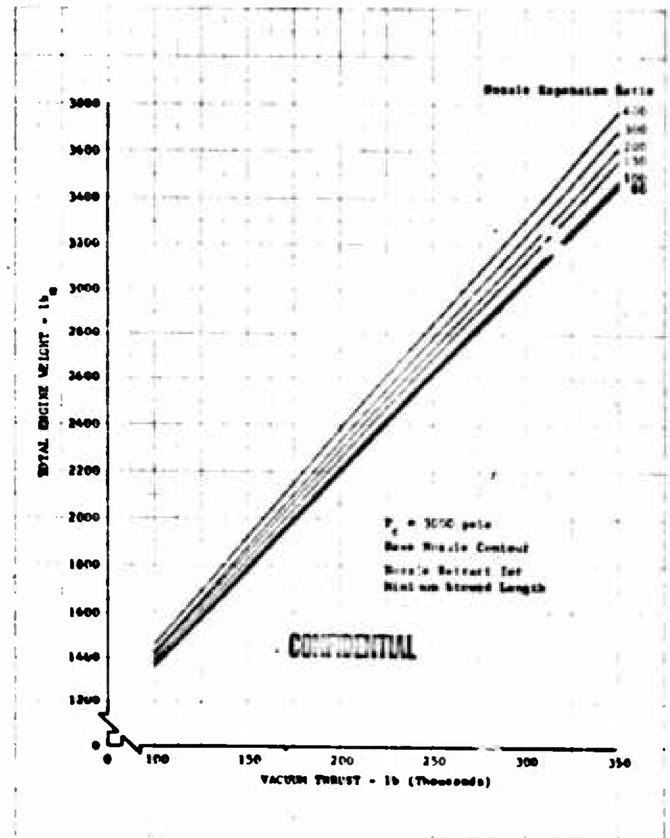


Figure 194. Engine Weight With Base Contour Two-Position Nozzle (ϵ_p = Minimum) DF 56354

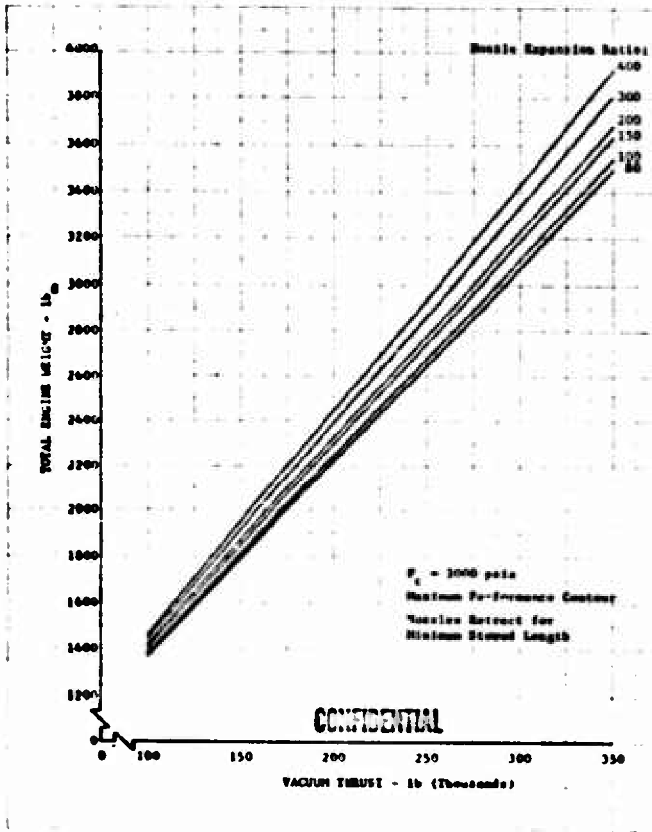


Figure 195. Engine Weight With Maximum Performance Contour Two-Position Nozzle (ϵ_p = Minimum) DF 56353

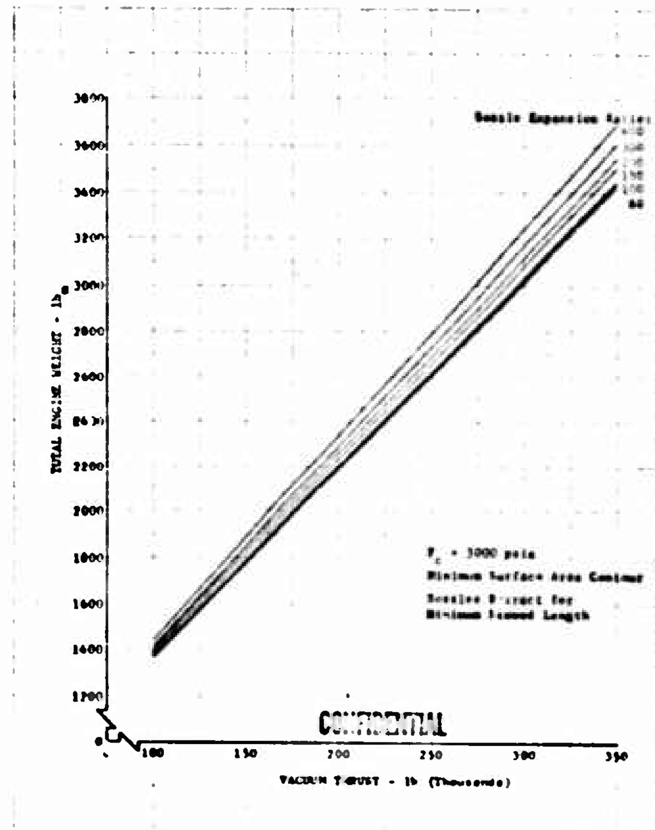


Figure 196. Engine Weight With Minimum Surface Area Contour, Two-Position Nozzle (ϵ_p = Minimum) DF 56355

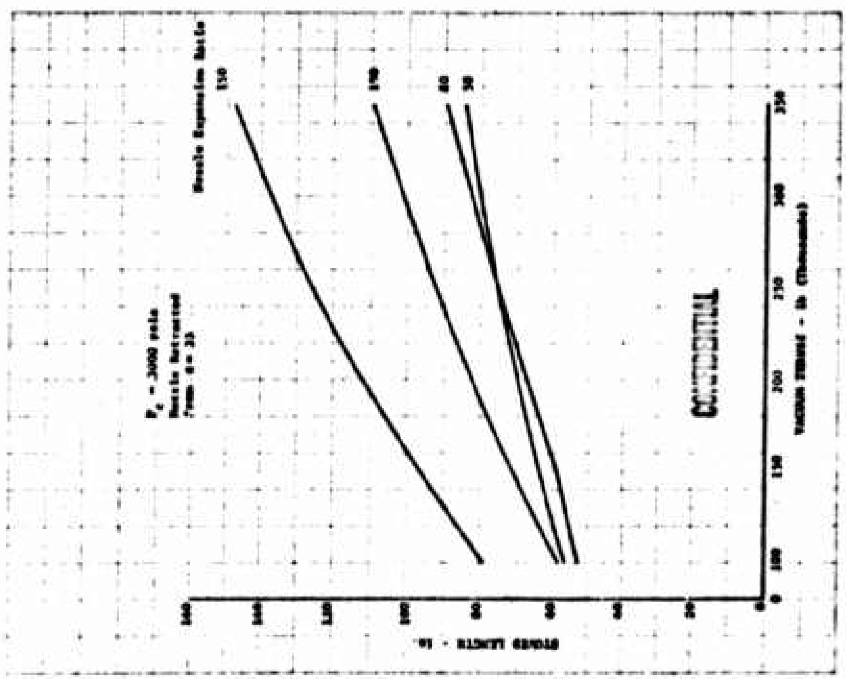


Figure 197. Stowed Length With Base Contour Two-Position Nozzle (ϵ_p = 35) DF 56239

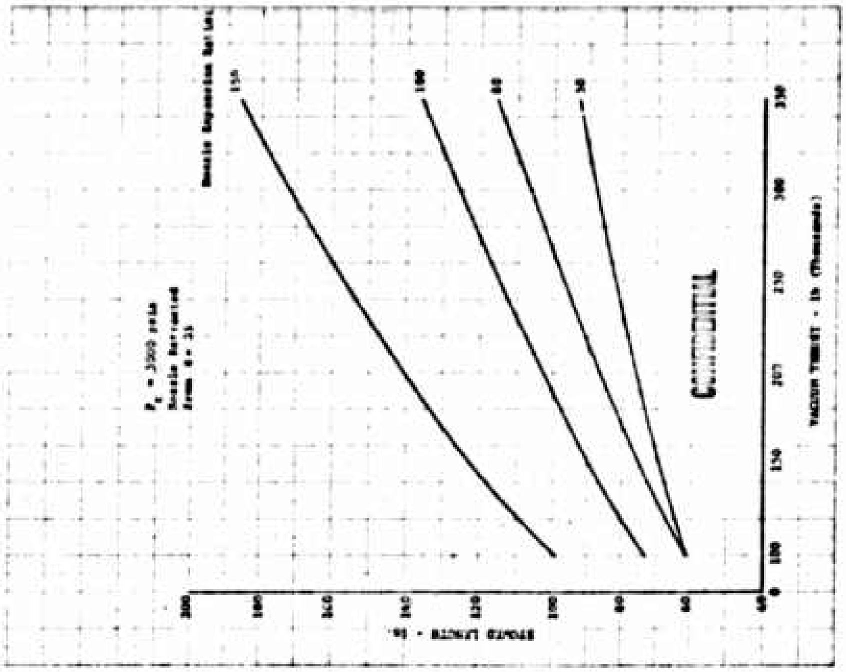


Figure 198. Stowed Length With Maximum Performance Contour Two-Position Nozzle (ϵ_p = 35) DF 56238

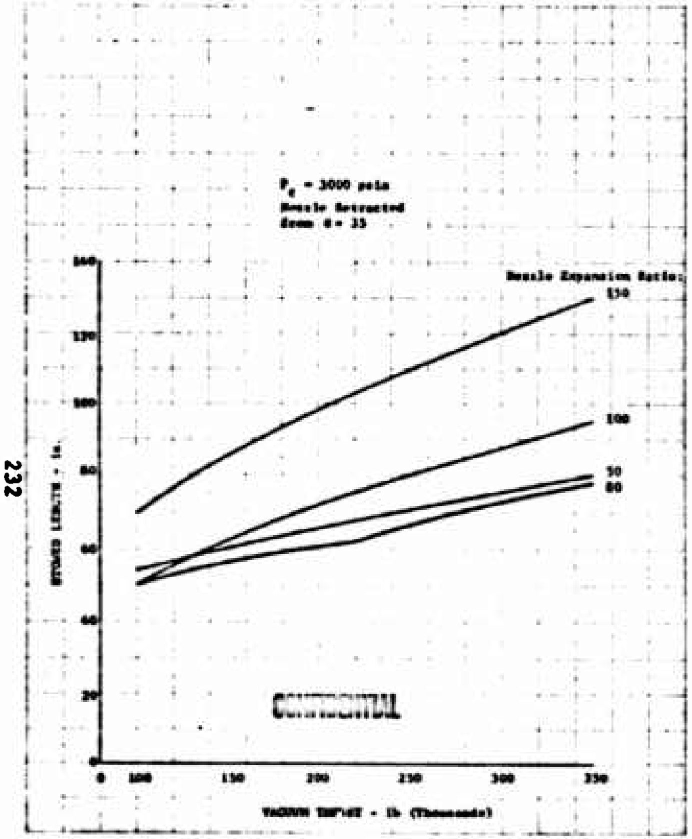


Figure 199. Stowed Length With Minimum Surface Area Contour Two-Position Nozzle (ϵ_p = 35) DF 56237

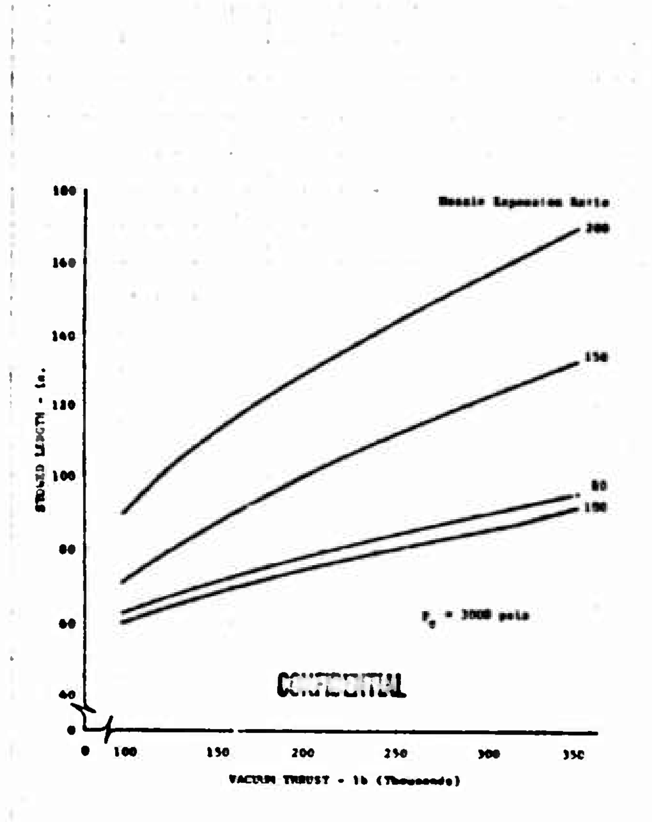


Figure 200. Stowed Length With Base Contour Two-Position Nozzle (ϵ_p = 50) DF 57275

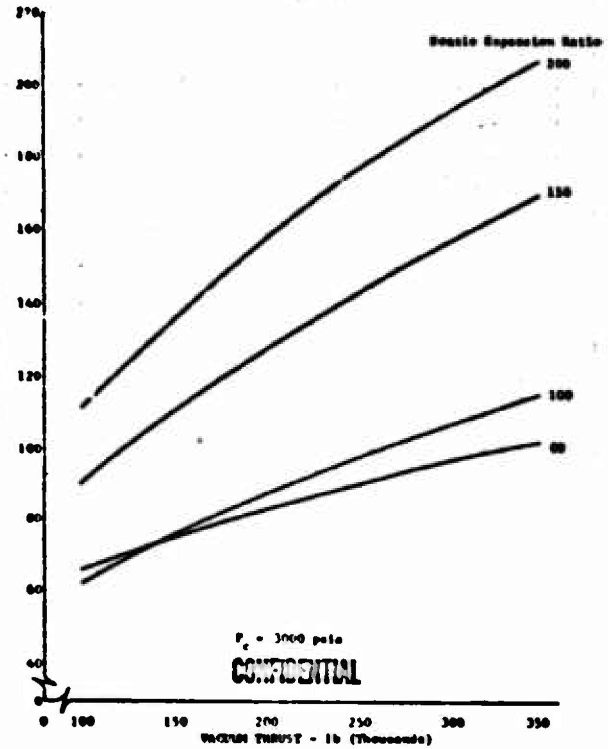


Figure 201. Stowed Length With Maximum Performance Contour Two-Position Nozzle (ϵ_p = 50) DF 57276

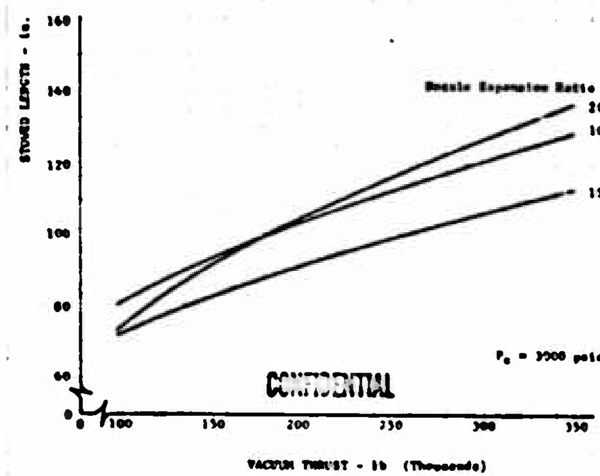


Figure 202. Stowed Length With Base Contour Two-Position Nozzle (ϵ_p = 80) DF 57277

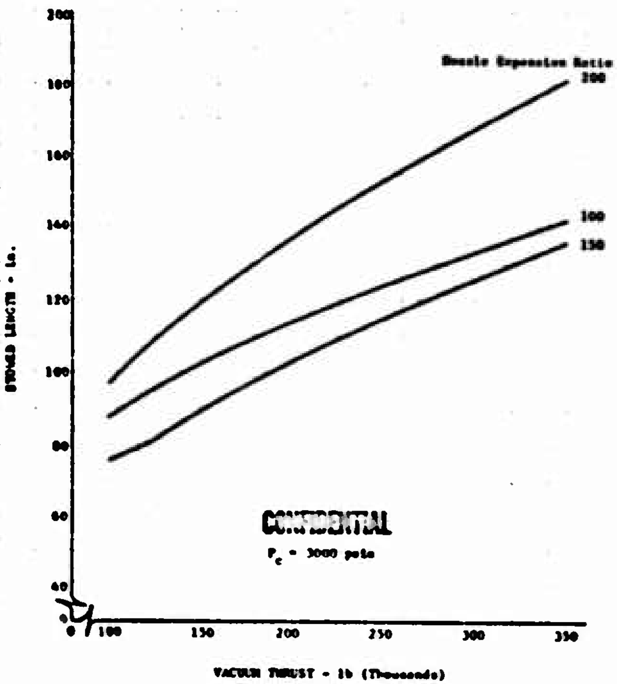


Figure 203. Stowed Length With Maximum Performance Contour Two-Position Nozzle (ϵ_p = 80) DF 57278

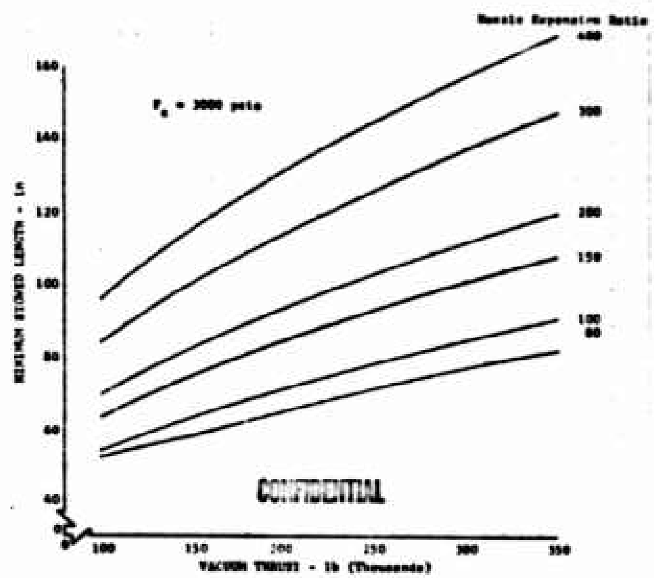


Figure 204. Minimum Stowed Length With Base Contour Two-Position Nozzle DF 56210

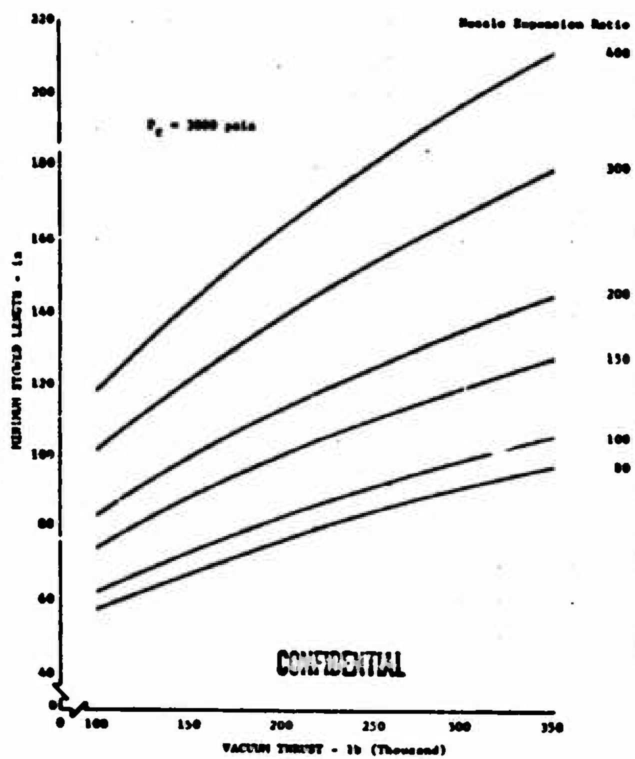


Figure 205. Minimum Stowed Length With Maximum Performance Contour Two-Position Nozzle DF 56259

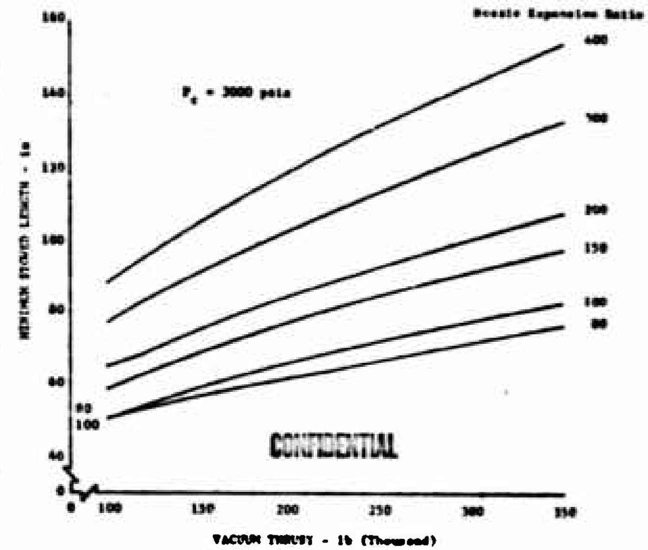


Figure 206. Minimum Stowed Length With Minimum Surface Area Contour Two-Position Nozzle DF 56258

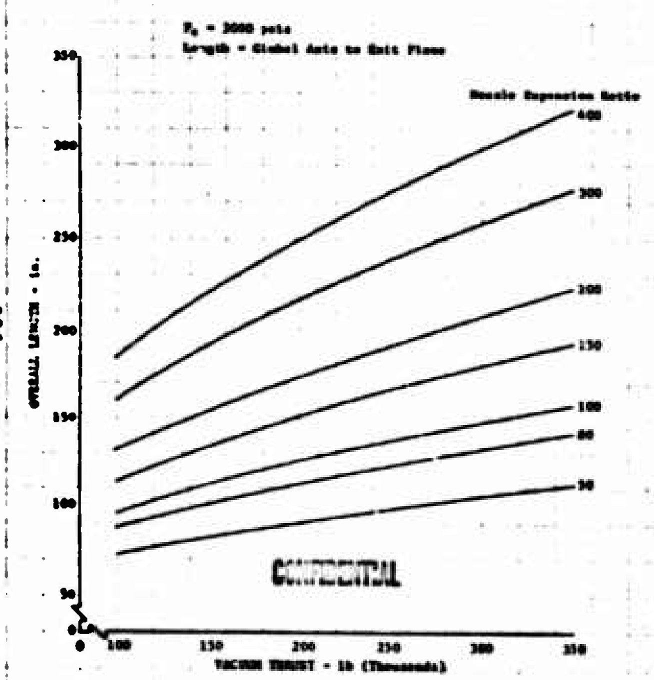


Figure 207. Overall Length With Base Contour Two-Position Nozzle DF 56212

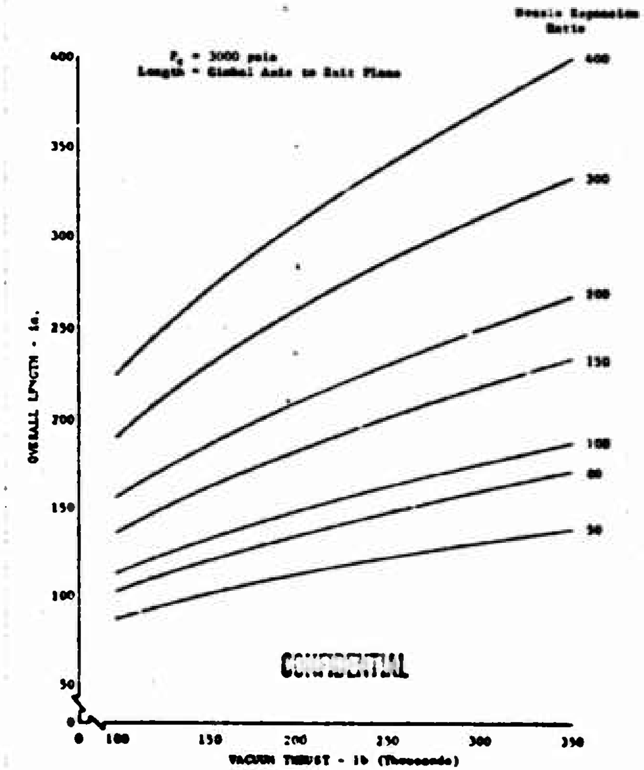


Figure 208. Overall Length With Maximum Performance Contour Two-Position Nozzle DF 56260

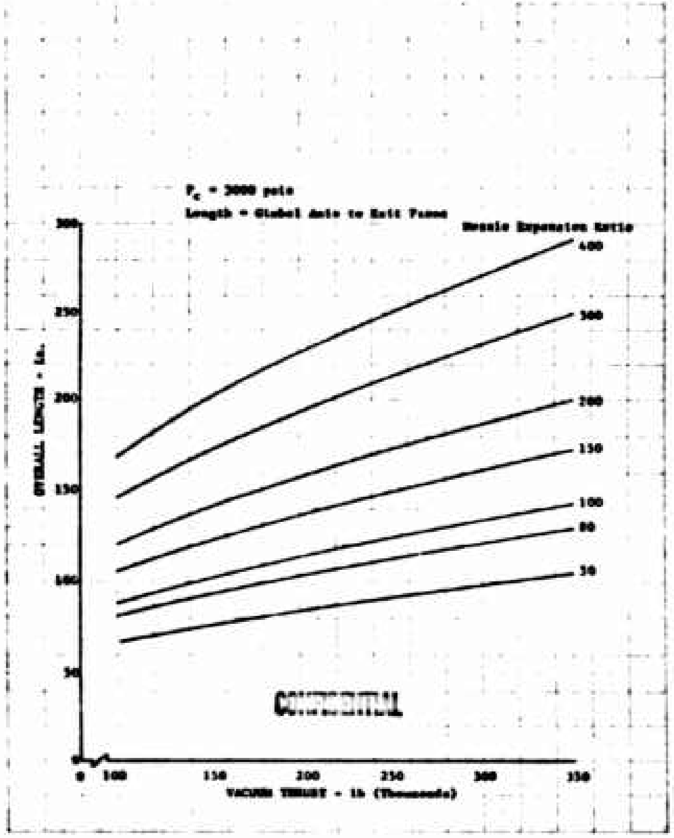


Figure 209. Overall Length With Minimum Surface Area Contour Two-Position Nozzle DF 56211

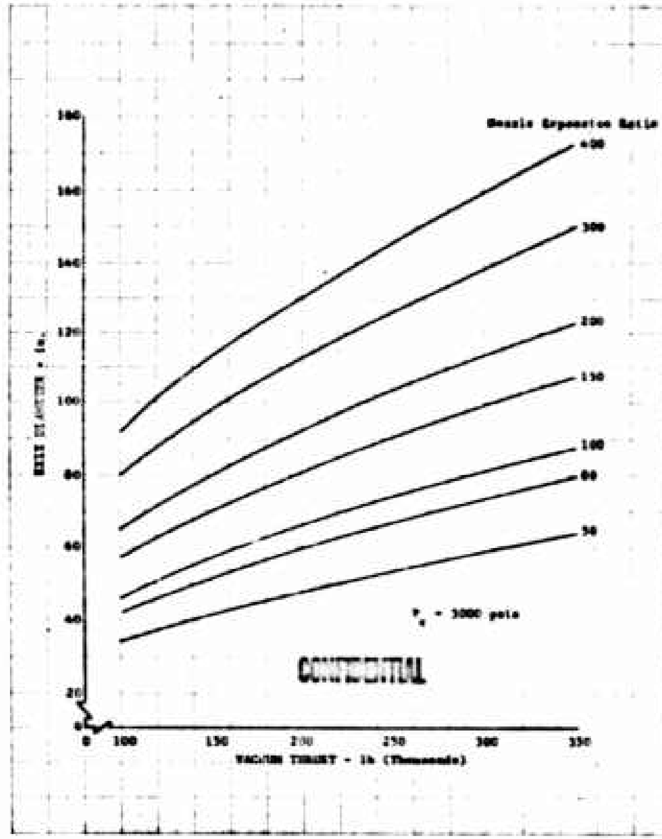


Figure 210. Exit Diameter (Two-Position Nozzle) DF 56257

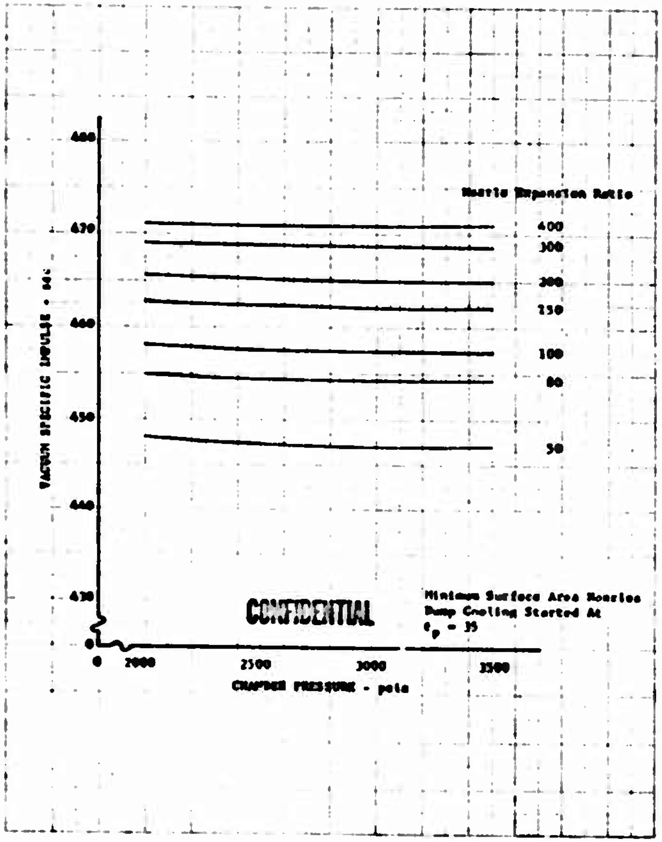


Figure 211. Vacuum Specific Impulse vs Chamber Pressure, 250K Module (Mixture Ratio of 5) DF 57246

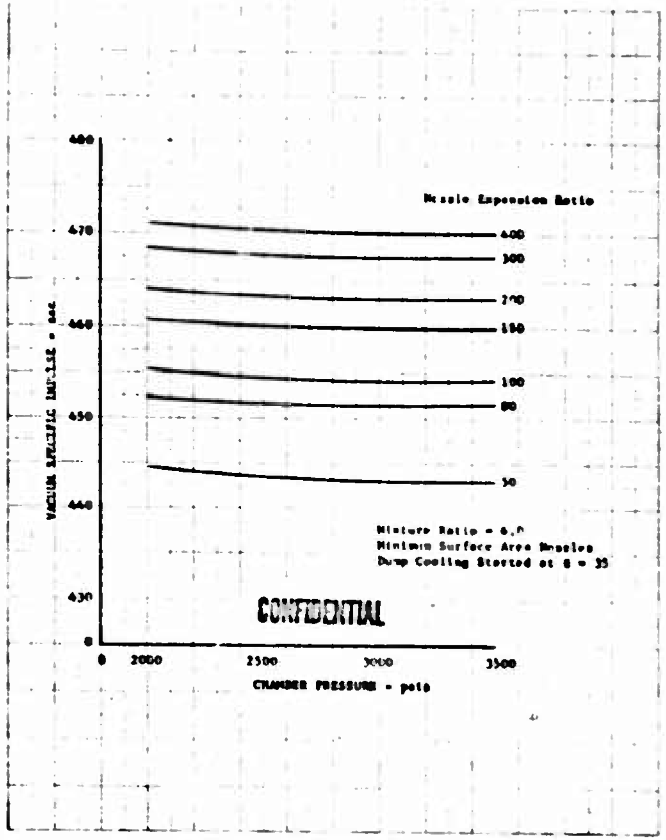


Figure 212. Vacuum Specific Impulse vs Chamber Pressure, 250K Module (Mixture Ratio of 6) DF 56221

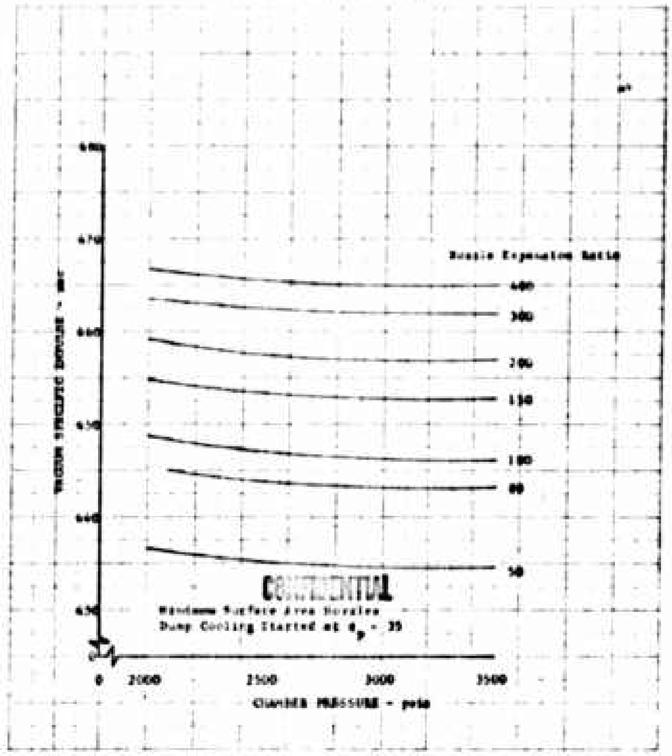


Figure 213. Vacuum Specific Impulse vs Chamber Pressure, 250K Module (Mixture Ratio of 7) DF 57247

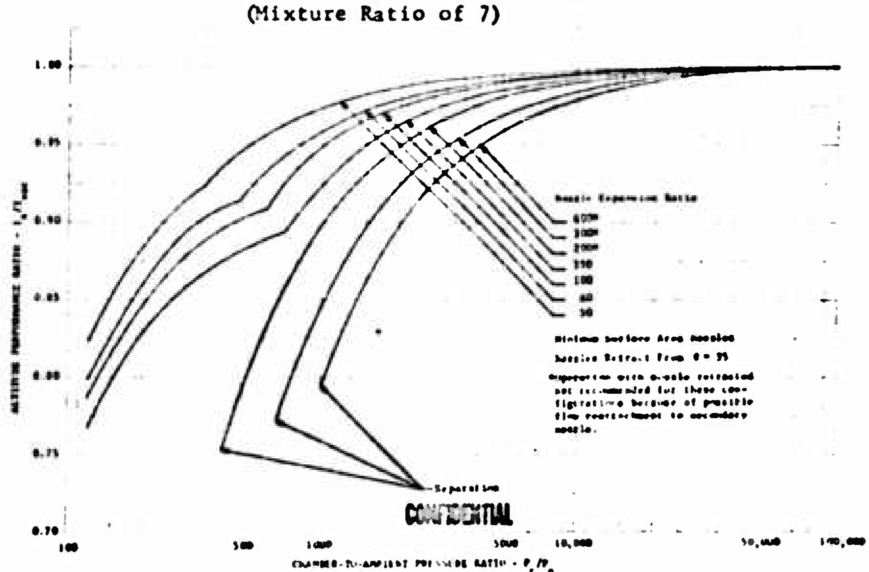


Figure 214. Two-Position Nozzle Altitude Performance at Mixture Ratio of 5.0 DF 56034

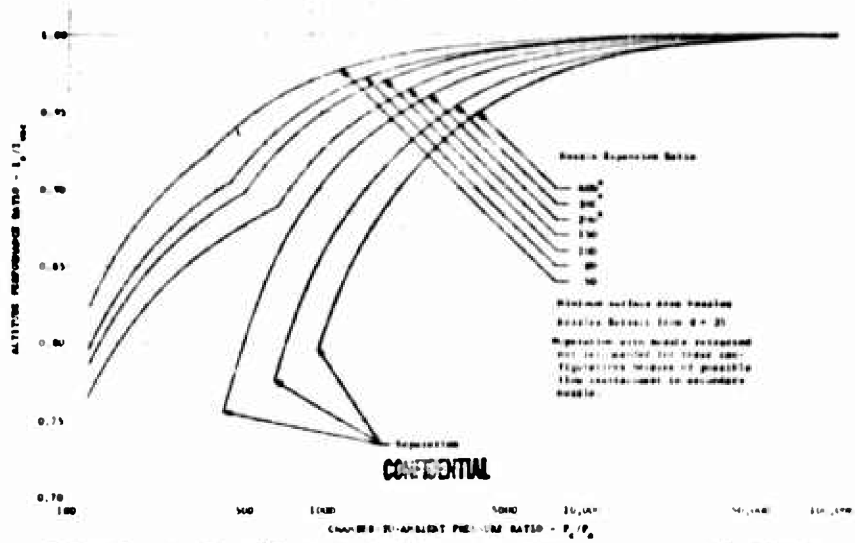


Figure 215. Two-Position Nozzle Altitude Performance at Mixture Ratio of 6.0 DF 56035

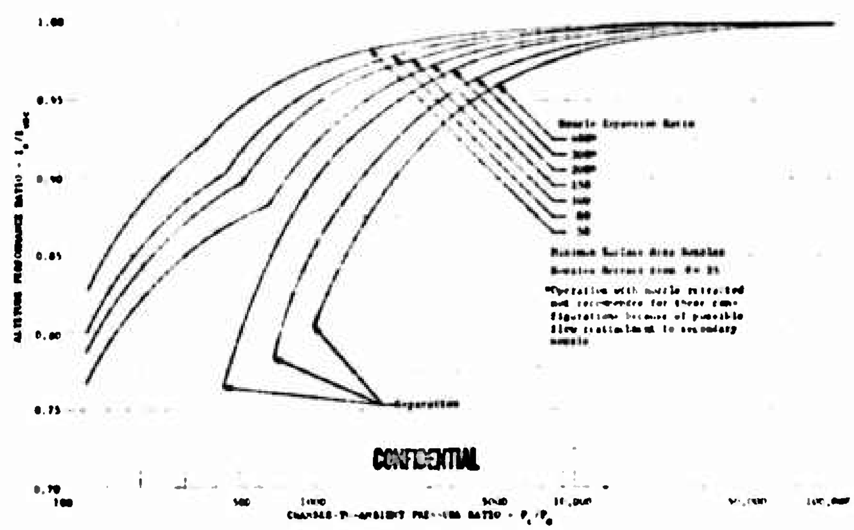


Figure 216. Two-Position Nozzle Altitude Performance at Mixture Ratio of 7.0 DF 56036

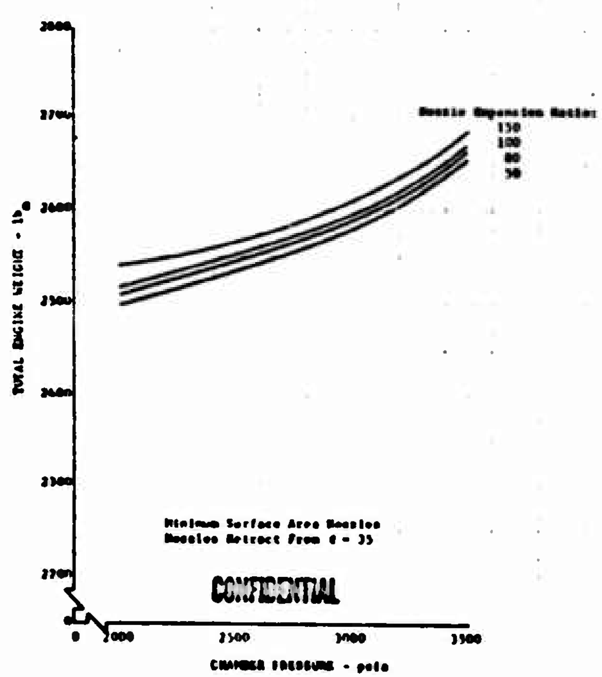


Figure 217. Weight vs Chamber Pressure, $\epsilon_p = 35$ DF 56350

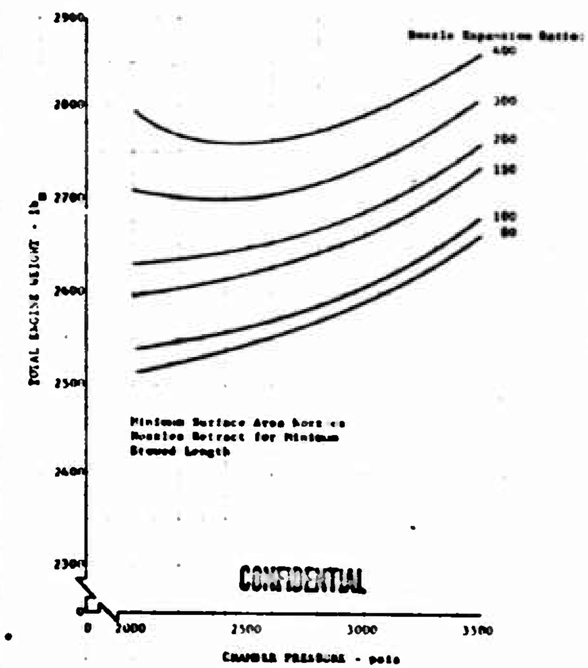


Figure 218. Weight vs Chamber Pressure, ϵ_p = Minimum DF 56344

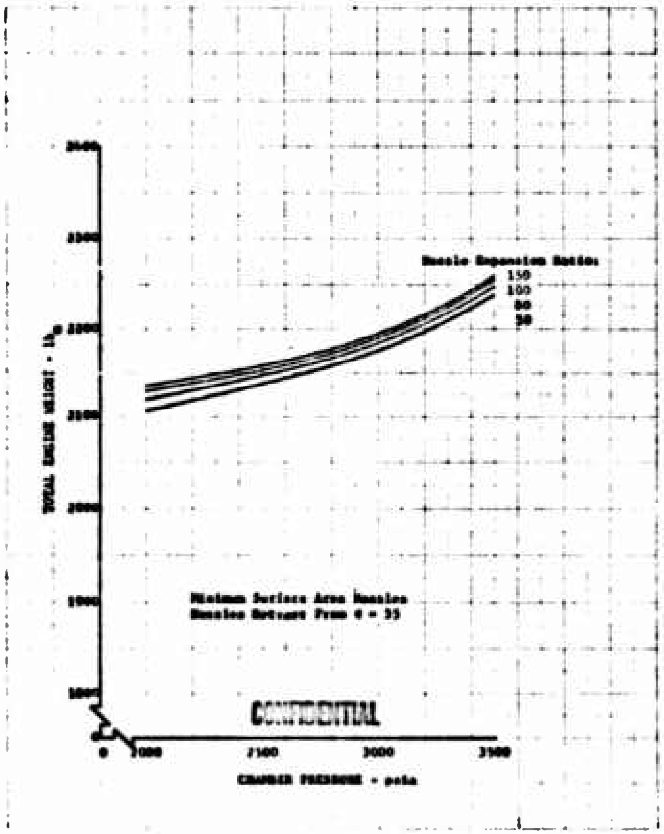


Figure 219. Engine Weight With Minimum Surface Area Contour Two-Position Nozzle, $\epsilon_p = 35$ (200K Thrust) DF 56349

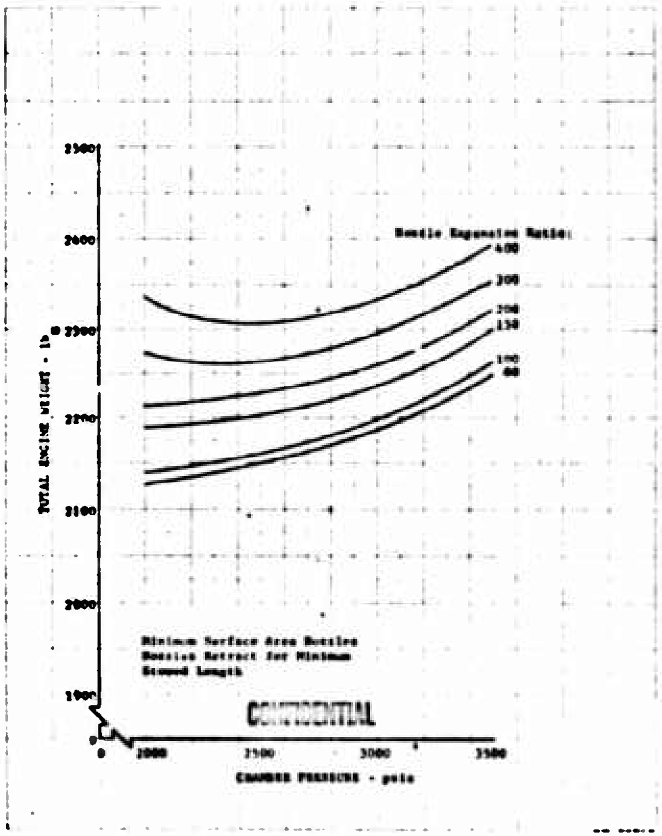


Figure 220. Engine Weight With Minimum Surface Area Contour Two-Position Nozzle, ϵ_p = Minimum (200K Thrust) DF 56343

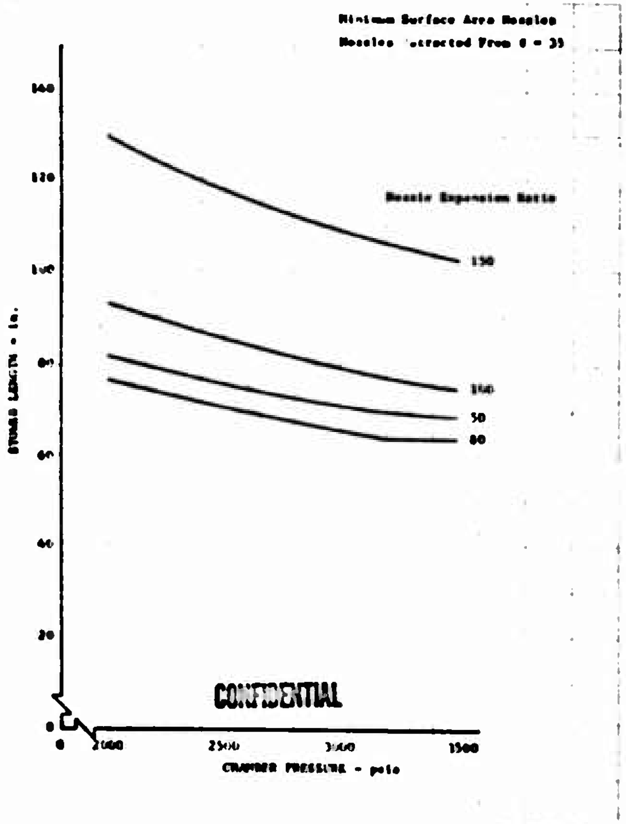


Figure 221. Stowed Length vs Chamber Pressure. ϵ_p = 35 DF 56227

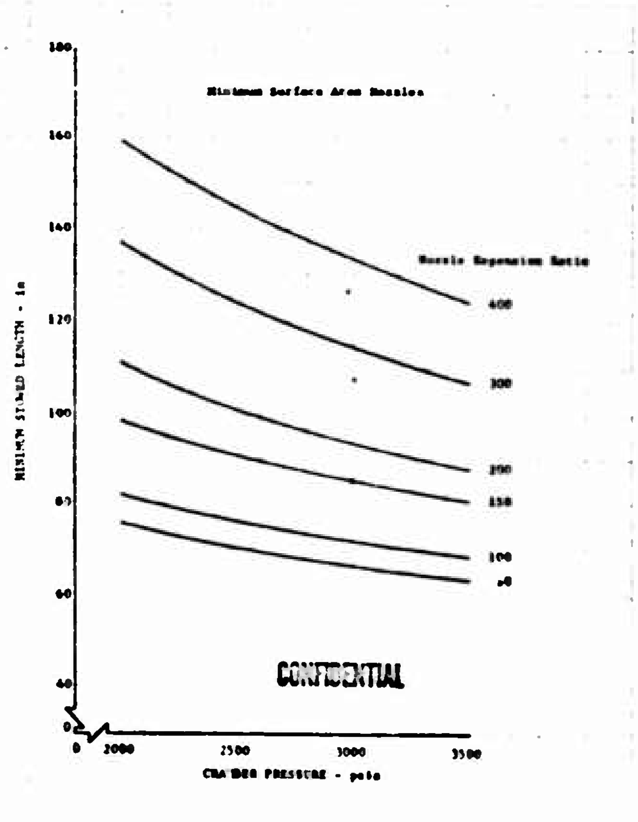


Figure 222. Minimum Stowed Length vs Chamber Pressure DF 56233

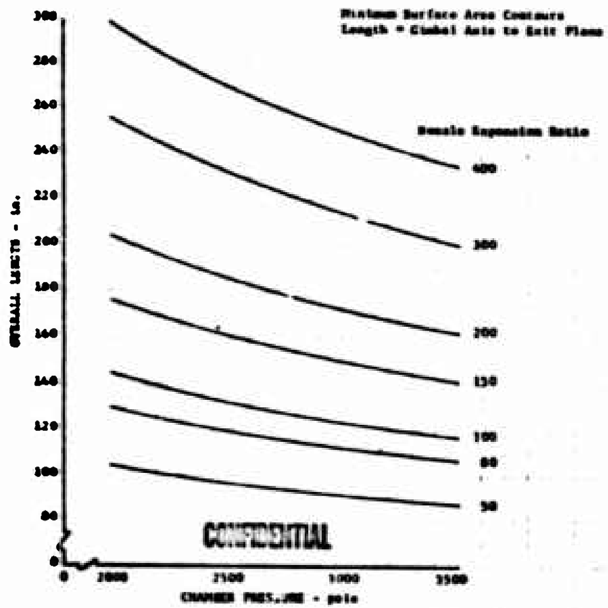


Figure 223. Overall Length vs Chamber Pressure DF 55800

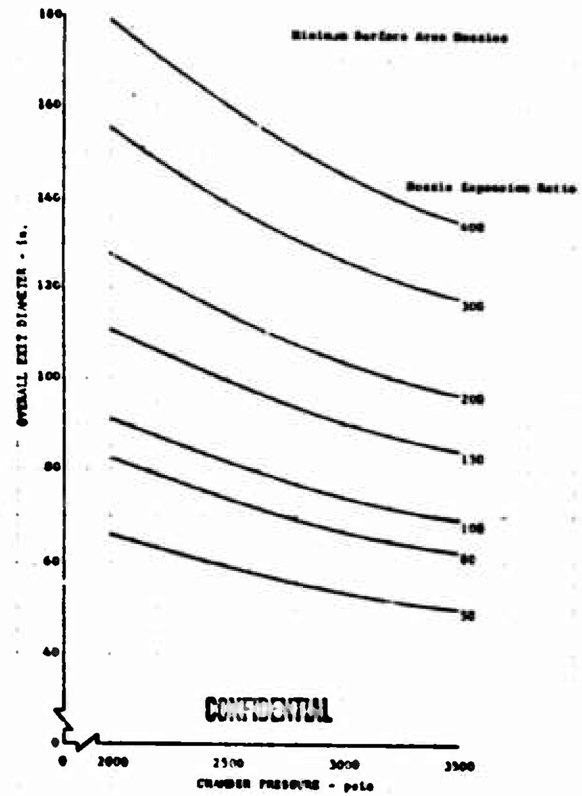


Figure 224. Overall Exit Diameter vs Chamber Pressure DF 56215

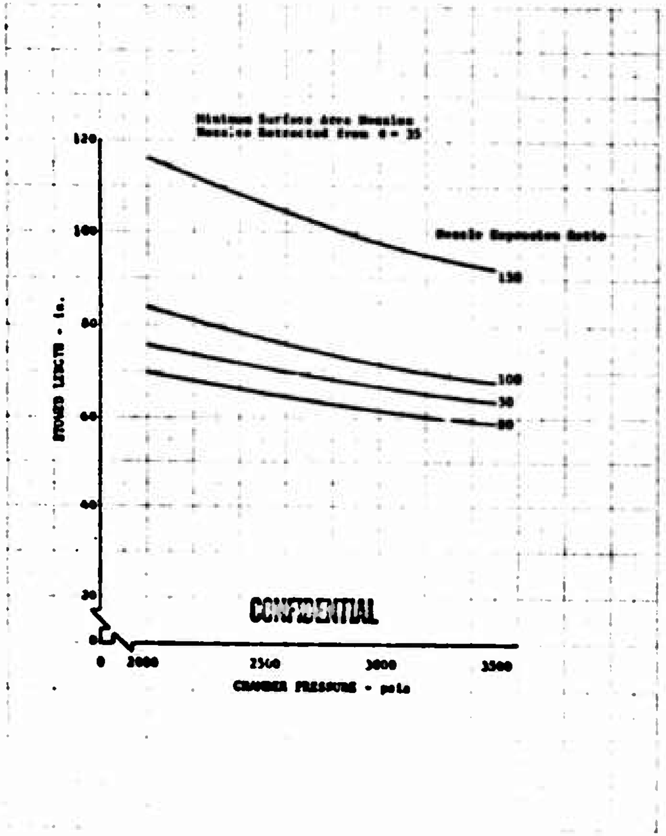


Figure 225. Stowed Length With Minimum Surface Area Contour Two-Position Nozzle, $\epsilon_p = 35$ (200K Thrust) DF 56228

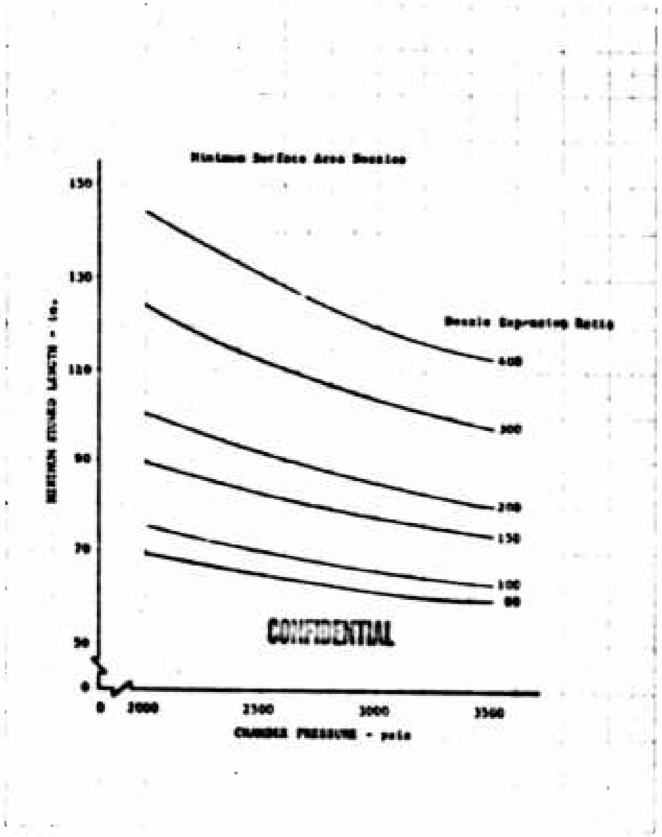


Figure 226. Minimum Stowed Length With Minimum Surface Area Contour Two-Position Nozzle (200K Thrust) DF 56234

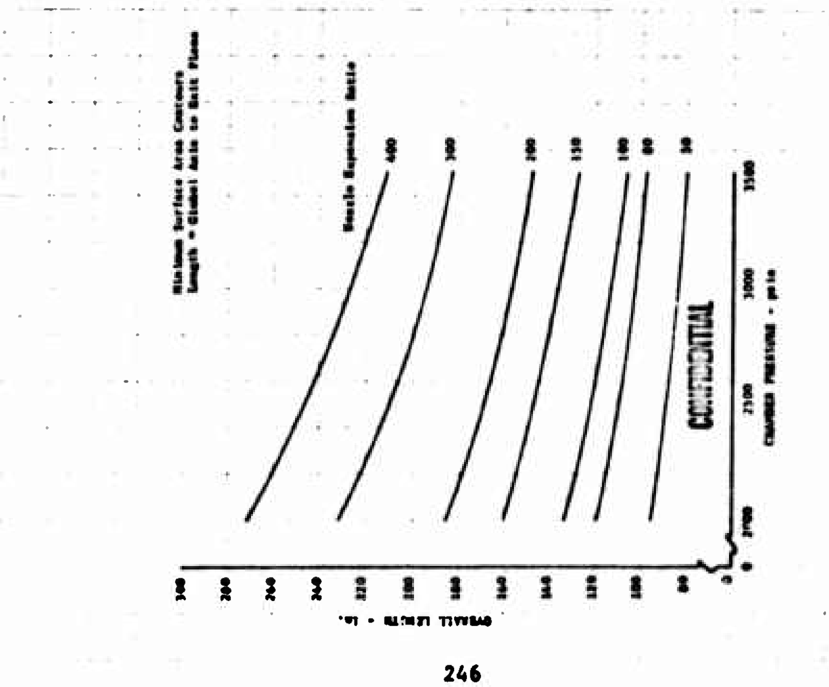


Figure 227. Overall Length With Minimum Surface Area Contour Two-Position Nozzle (200K Thrust) DF 55801

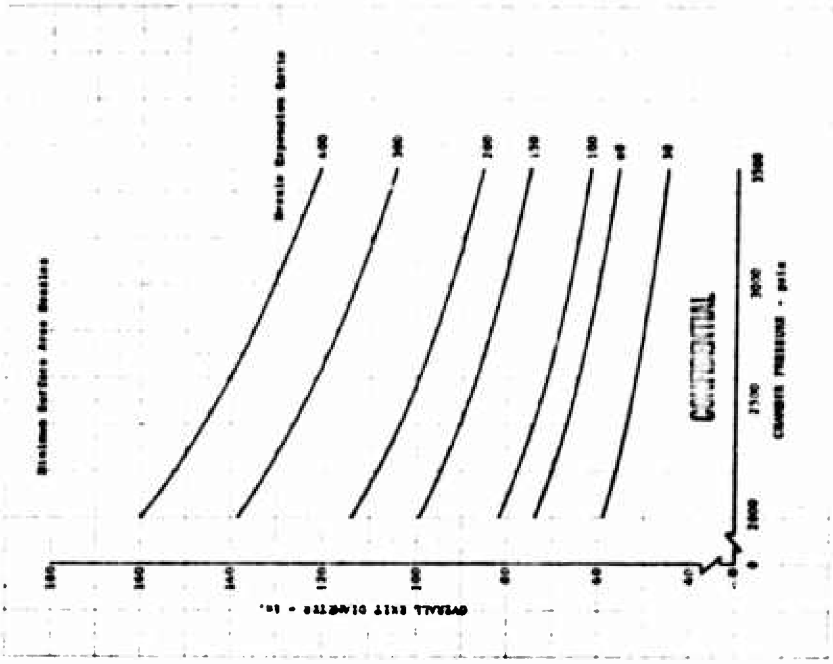


Figure 228. Overall Exit Diameter vs Chamber Pressure (200K Thrust) DF 56216

B. FIXED POSITION LIGHTWEIGHT NOZZLE

(C) The data presented in this section are for the high pressure staged combustion engine utilizing a fixed lightweight nozzle. The basic engine design requirements and features are identical to those discussed in the previous paragraph except that the two-position nozzle feature is not provided. Data for the fixed lightweight nozzle are given for a chamber pressure of 3000 psia.

1. Performance

(U) Delivered vacuum impulse for the fixed lightweight nozzle is identical to that of the two-position lightweight nozzle and can be obtained from figures 163 through 168.

(C) Performance from sea level to 200,000 feet for the fixed nozzle is shown as a ratio of delivered specific impulse to vacuum impulse plotted as a function of altitude. These data are presented in figures 229 through 231 for a mixture ratio of 5.0, 6.0, and 7.0.

(U) Using the above ratio curves, the delivered specific impulse at other than vacuum conditions can be calculated. The delivered specific impulse at any altitude up to 200,000 ft (above 200,000 ft is considered vacuum conditions) is calculated by:

$$I_{s_{alt}} = \frac{I_{s_{alt}}}{I_{vac}} I_{vac}$$

where:

$I_{s_{alt}}$ = Delivered specific impulse at the altitude of interest

$I_{s_{alt}}/I_{vac}$ = Ratio of altitude to vacuum performance for the altitude and engine conditions of interest

I_{vac} = Delivered vacuum specific impulse for the engine conditions and secondary area ratio of interest

also:

$$F_{alt} = \frac{I_{s_{alt}}}{I_{vac}} F_{vac}$$

2. Weight

(U) Weight for engines using a fixed lightweight nozzle is presented in figures 232 through 234.

3. Dimensional Data

(U) The overall nozzle diameters are presented in figure 235 and the total engine lengths presented in figures 236 through 238. These data are the same as that used for the two-position nozzle but are repeated to extend the data to lower expansion ratios.

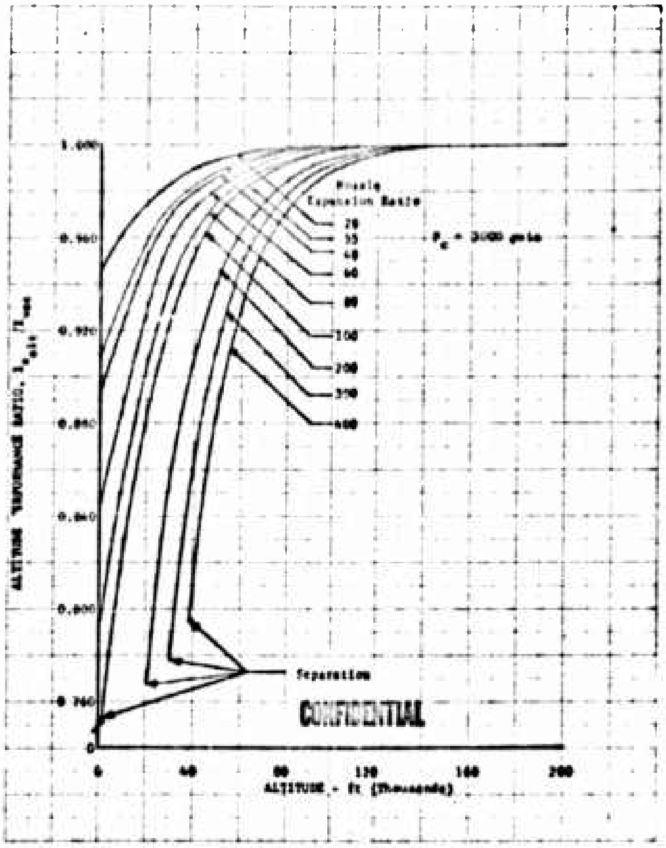


Figure 229. Fixed Lightweight Nozzle Altitude Performance at Mixture Ratio of 5.0 DF 59929

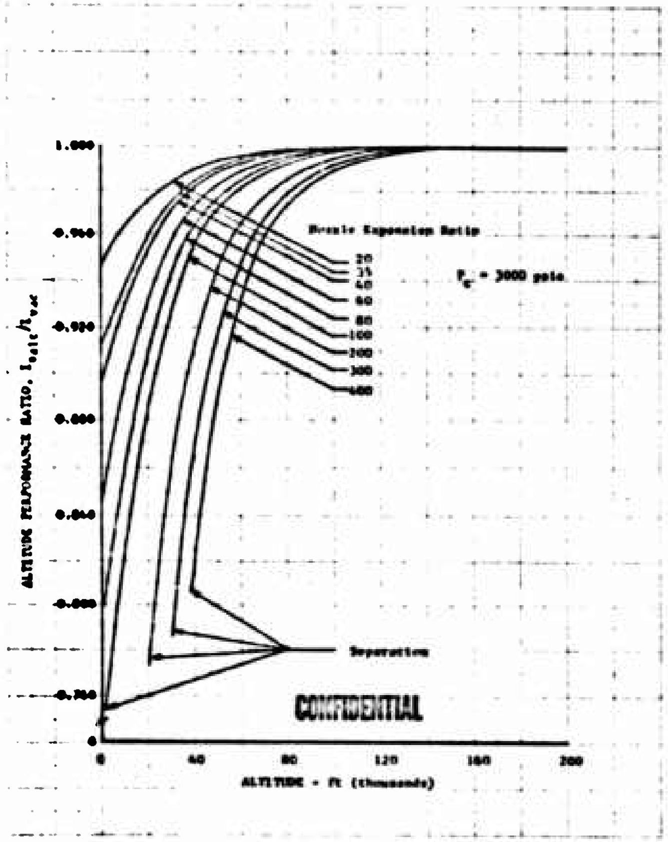


Figure 230. Fixed Lightweight Nozzle Altitude Performance at Mixture Ratio of 6.0 DF 59930

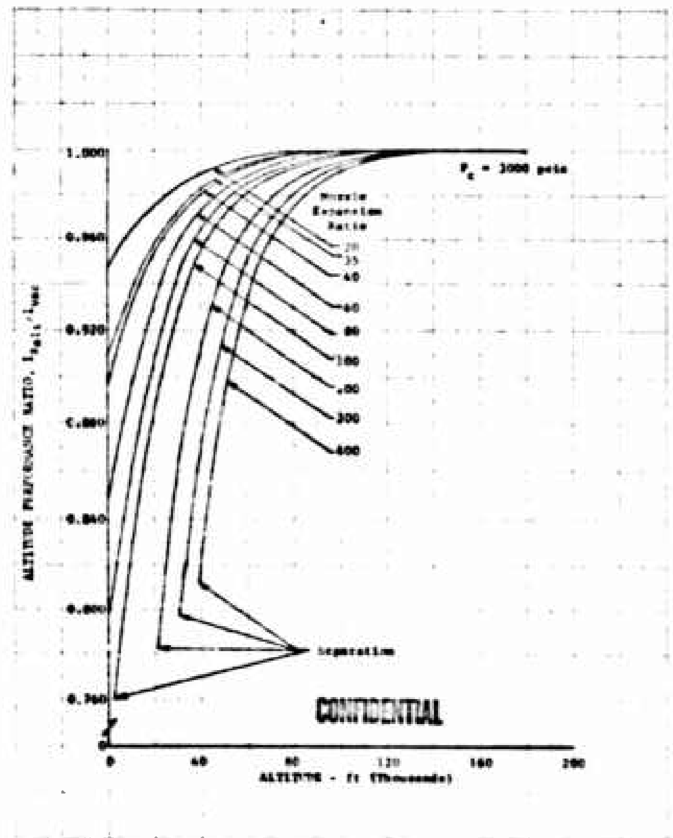


Figure 231. Fixed Lightweight Nozzle Altitude Performance at Mixture Ratio of 7.0 DF 59931

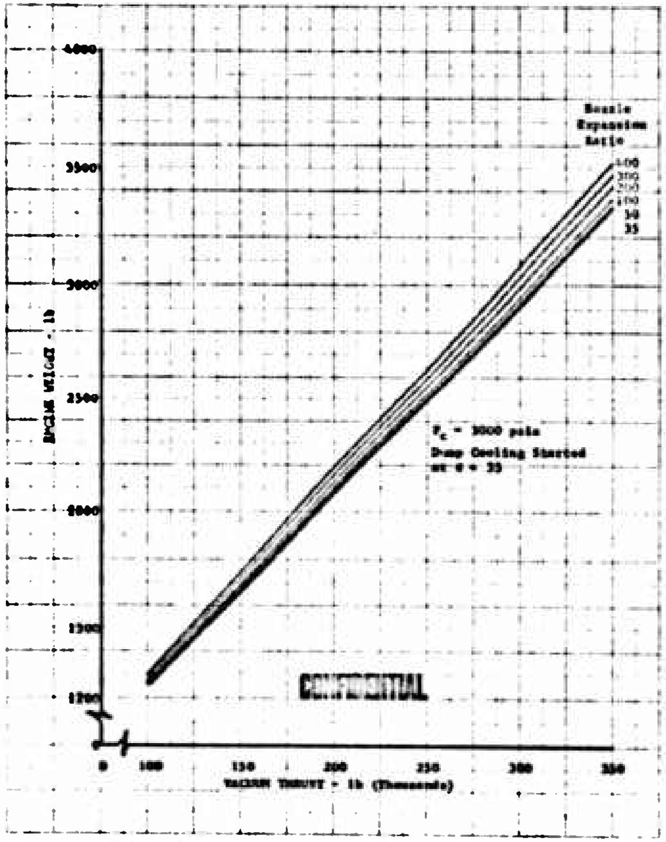


Figure 232. Engine Weight With Base Contour Fixed Lightweight Nozzle DF 59936

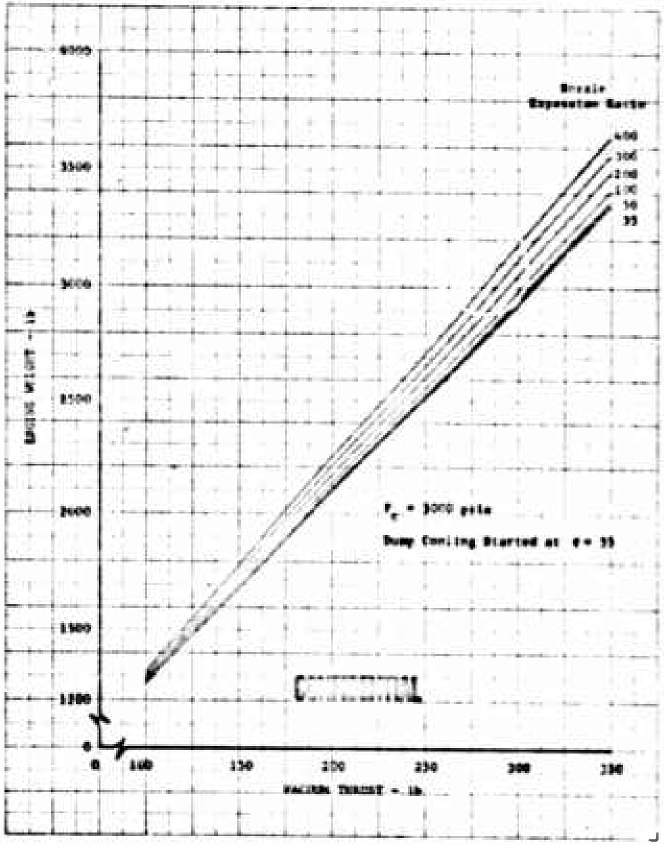


Figure 233. Engine Weight With Maximum Performance Contour Fixed Lightweight Nozzle DF 59937

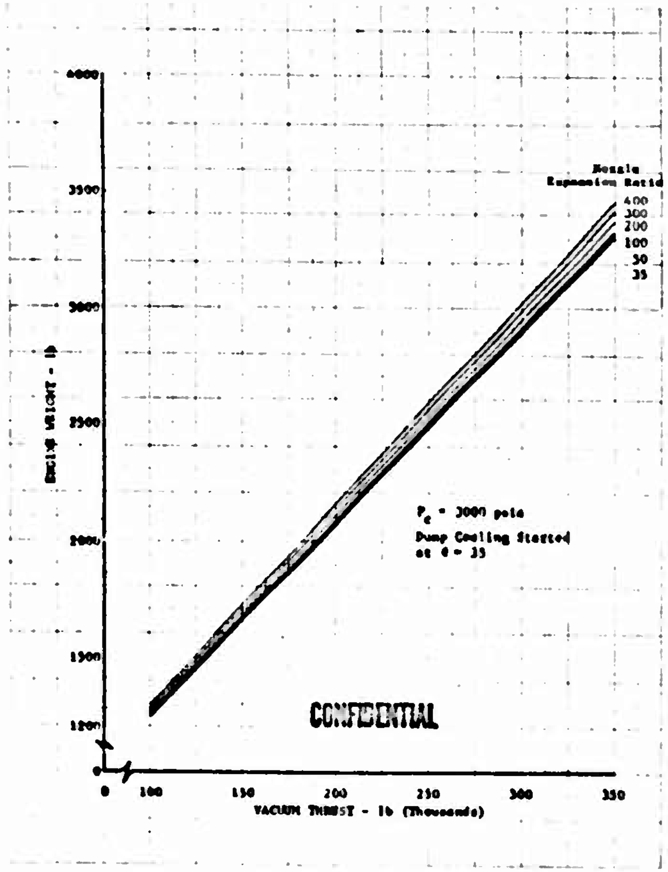


Figure 234. Engine Weight With Minimum Surface Area Contour Fixed Lightweight Nozzle DF 59938

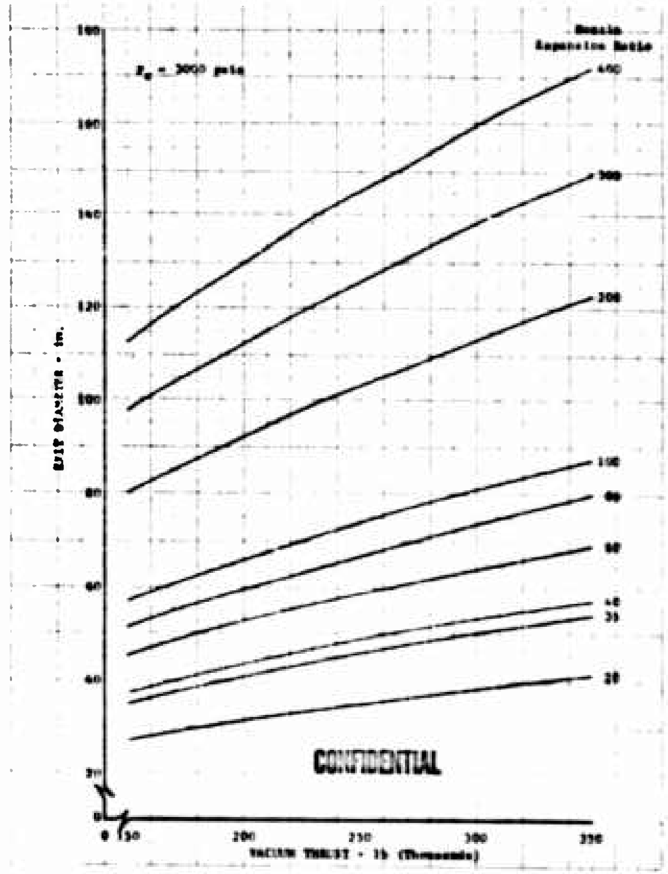


Figure 235. Exit Diameter (Fixed Lightweight Nozzle) DF 59932

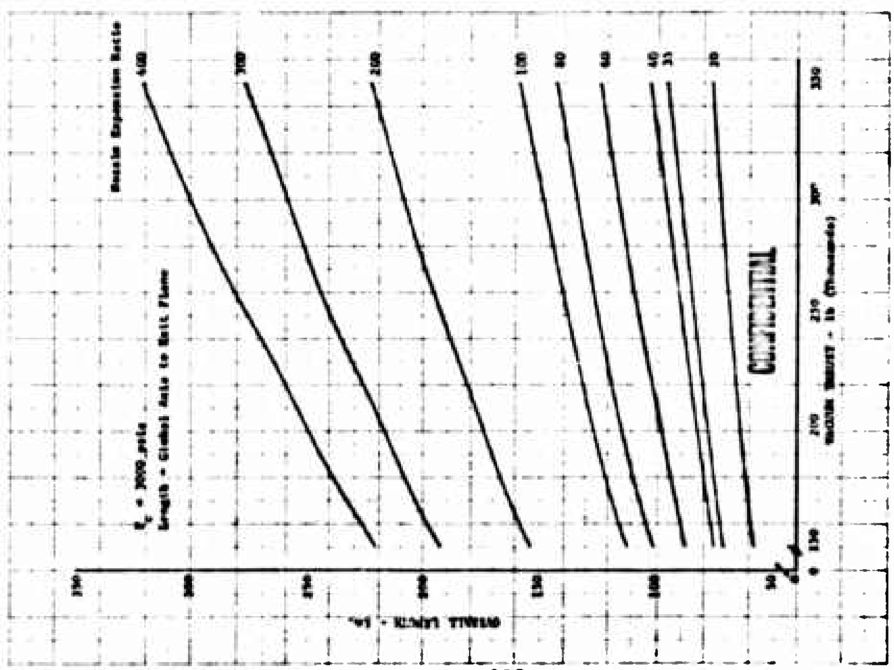


Figure 236. Overall Length With Base Contour Fixed Lightweight Nozzle DF 59934

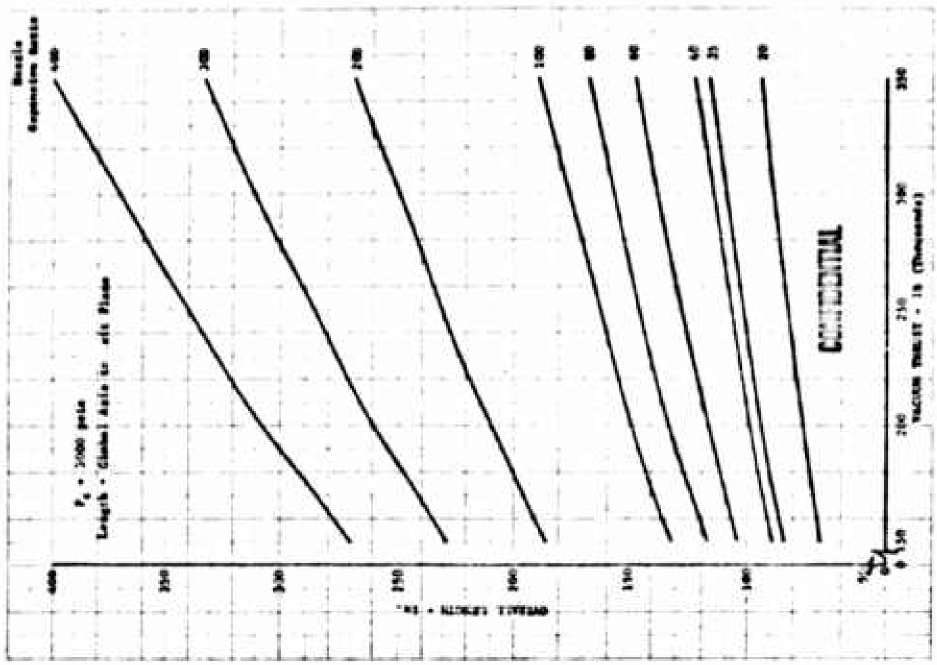


Figure 237. Overall Length With Maximum Performance Contour Fixed Lightweight Nozzle DF 59935

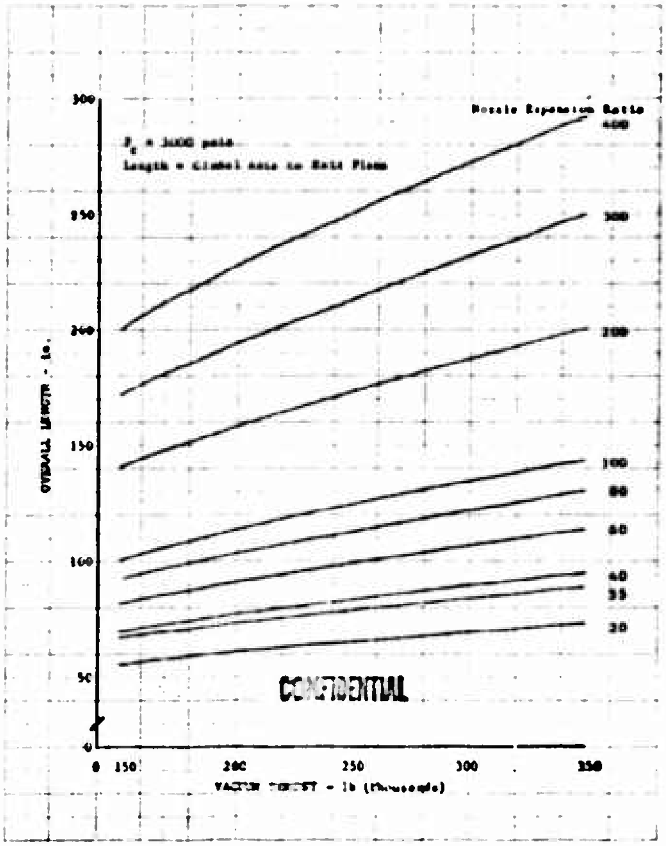


Figure 238. Overall Length With Minimum Surface Area Contour Fixed Lightweight Nozzle DF 59933

C. ENGINE WITH FIXED REGENERATIVE NOZZLE

(C) Data presented in this paragraph are for high-pressure staged-combustion engines using fixed regeneratively cooled nozzles. These data are based on the same engine requirements and characteristics previously referenced except for the use of a fixed regeneratively cooled nozzle. Data are given for a chamber pressure of 3000 psia. Data are limited to a vacuum thrust level of 250K (350K) or less.

1. Performance

(U) Delivered vacuum performance is presented for thrust levels of 250K and 350K in figures 239 through 244. The ratio of delivered specific impulse at any altitude from sea level to 200,000 feet to delivered vacuum impulse is the same as that given for the fixed lightweight nozzle in figures 229 through 231.

3. Dimensional Data

(U) The dimensional data are the same as those given for the fixed lightweight nozzle. Overall engine length is shown in figures 236 through 238 and nozzle exit diameter is given in figure 235.

2. Weight

(U) The total weights for engines using fixed fully regeneratively cooled nozzles are given in figures 245 through 247.

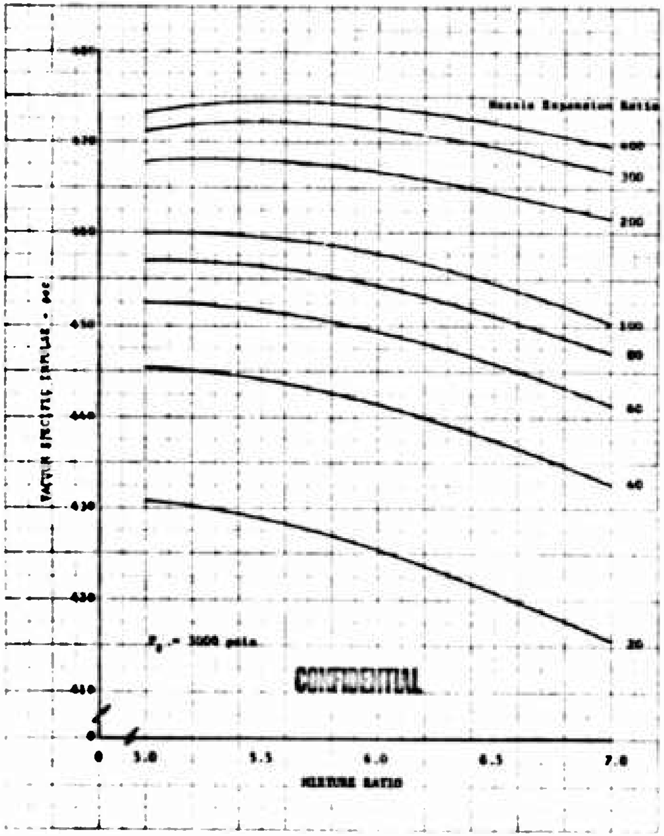


Figure 239. Vacuum Specific Impulse vs Mixture Ratio, Base Contour Fixed Regeneratively Cooled Nozzle (250K Module) DF 59939

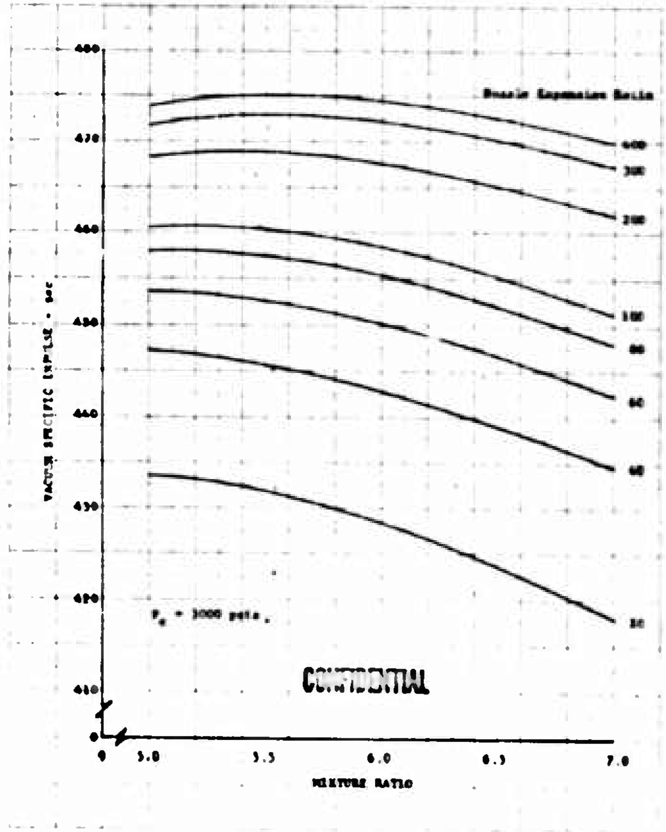


Figure 240. Vacuum Specific Impulse vs Mixture Ratio, Maximum Performance Contour Fixed Regeneratively Cooled Nozzle (250K Module) DF 59940

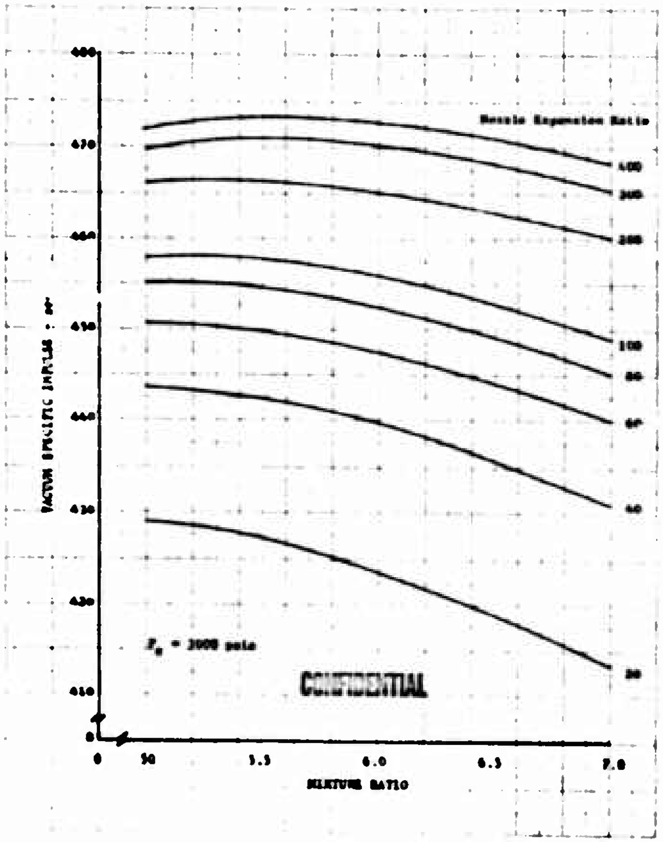


Figure 241. Vacuum Specific Impulse vs Mixture Ratio, Minimum Surface Area Contour Fixed Regeneratively Cooled Nozzle (250K Module) DF 59941

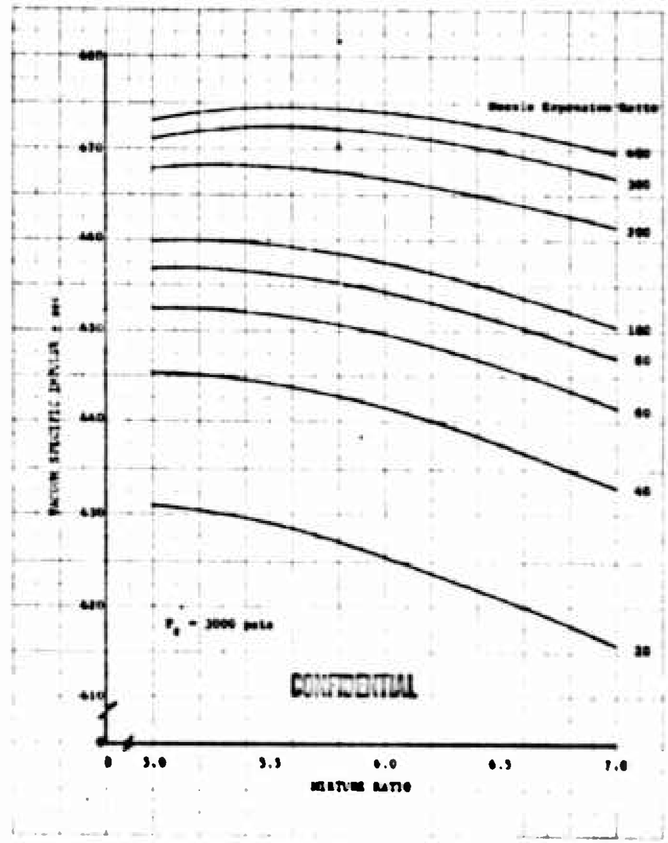


Figure 242. Vacuum Specific Impulse vs Mixture Ratio, Base Contour Fixed Regeneratively Cooled Nozzle (350K Module) DF 59942

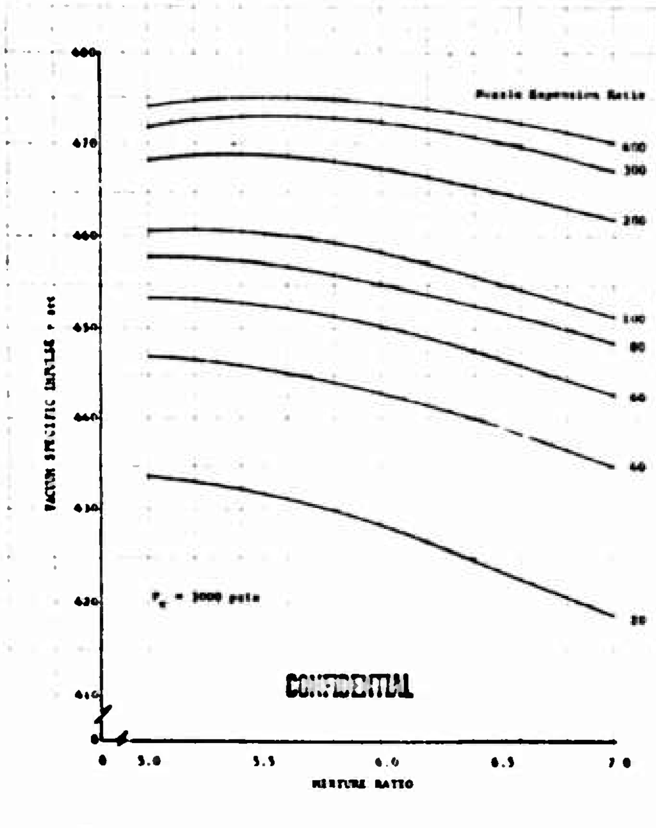


Figure 243. Vacuum Specific Impulse vs Mixture Ratio, Maximum Performance Contour Fixed Regeneratively Cooled Nozzle (350K Module) DF 59943

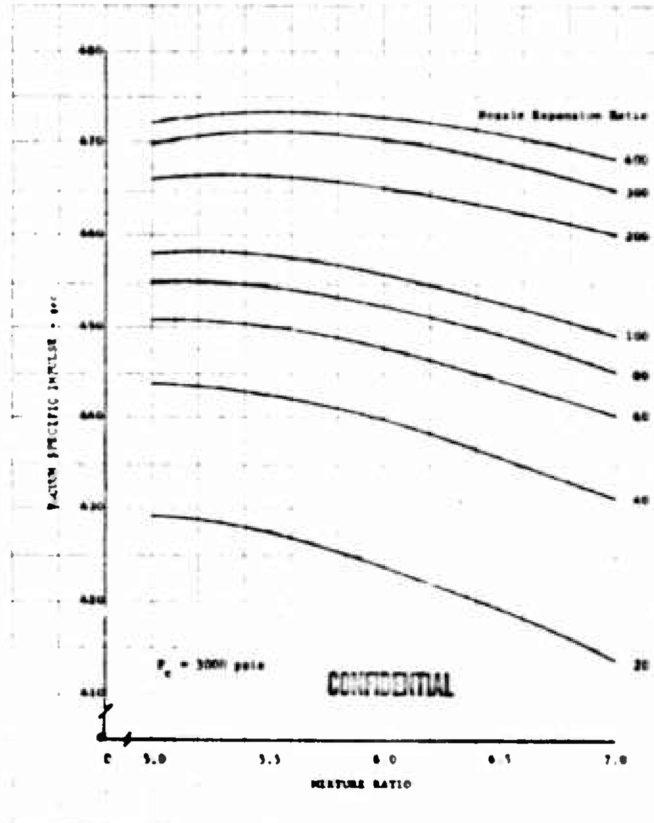


Figure 244. Vacuum Specific Impulse vs Mixture Ratio, Minimum Surface Area Contour Fixed Regeneratively Cooled Nozzle (350K Module) DF 59944

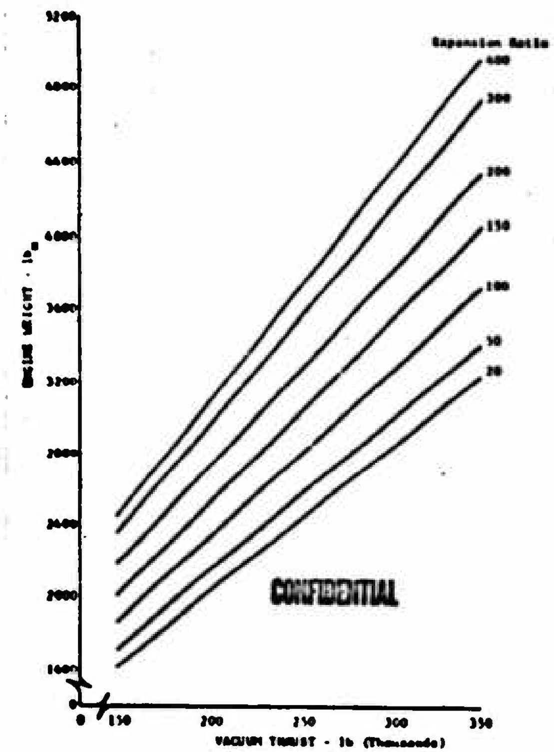


Figure 245. Engine Weight for Base Contour Fixed Regeneratively Cooled Nozzle DF 57068

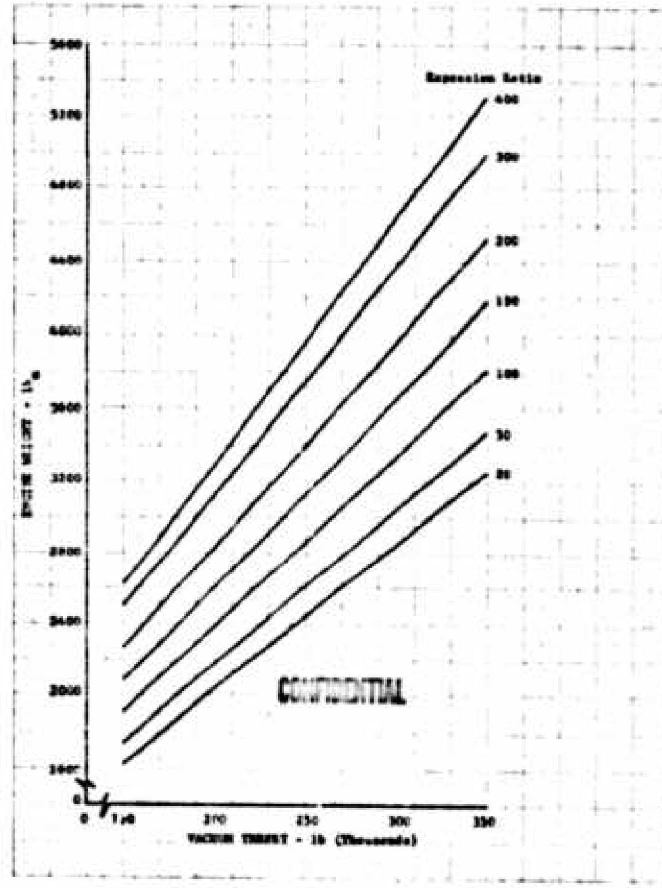


Figure 246. Engine Weight for Maximum Performance Contour Fixed Regeneratively Cooled Nozzle DF 57072

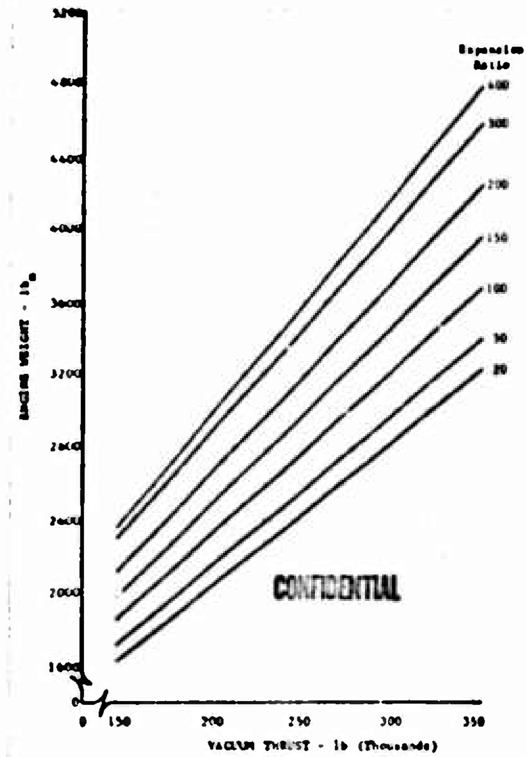


Figure 247. Engine Weight for Minimum Surface Area Contour Fixed Regeneratively Cooled Nozzle DF 57064

APPENDIX III
COMMON MODULE ENGINE DATA

(U) This appendix presents engine performance, weight, and size data for the specific 250K and 350K common module engines selected in the applications study. Engine data are given as a function of nozzle expansion ratio and contour with the fixed power package. These data are based on two-position lightweight nozzles.

A. PERFORMANCE

(U) The delivered vacuum specific impulse and sea level impulse is presented in figures 248 through 252 for the 250K common module and in figures 253 through 257 for the 350K common module.

B. WEIGHT

(U) The engine weight is given in figures 258 and 259.

C. DIMENSIONAL DATA

(C) The nozzle exit diameter is shown in figure 260. The envelope diameter of the turbomachinery package is 65 inches for the 250K engine and 80 inches for the 350K engine. The overall length is presented in figures 261 and 262. The stowed engine length for nozzles retracting from an expansion ratio of 35 are given in figures 263 and 264. The minimum stowed length is given in figures 265 and 266. The break in slope is caused by the point where the secondary nozzle can be retracted over the turbopump envelope.

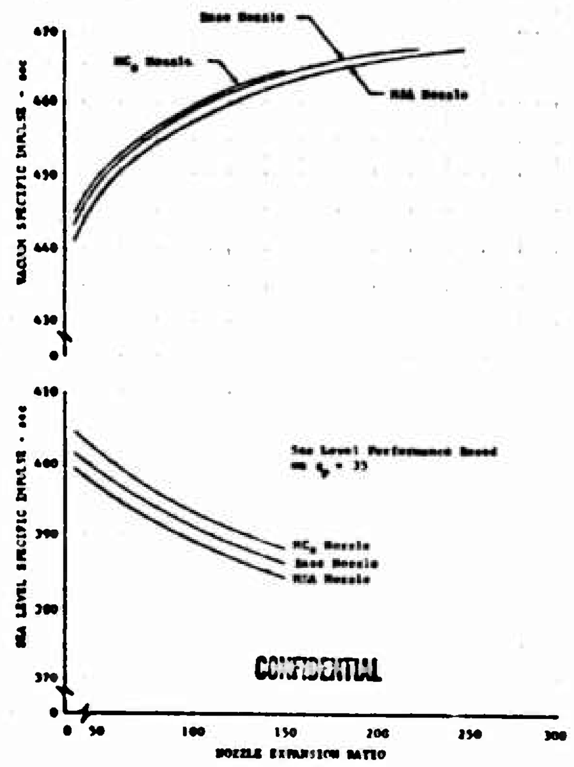


Figure 248. Sea Level and Vacuum Specific Impulse, 250K Common Module, Mixture Ratio of 5.0 DF 59945

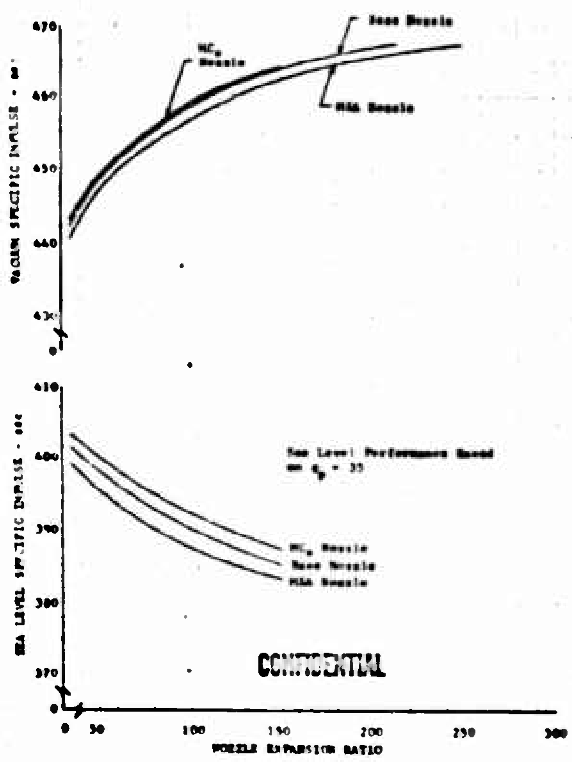


Figure 249. Sea Level and Vacuum Specific Impulse, 250K Common Module, Mixture Ratio of 5.5 DF 59946

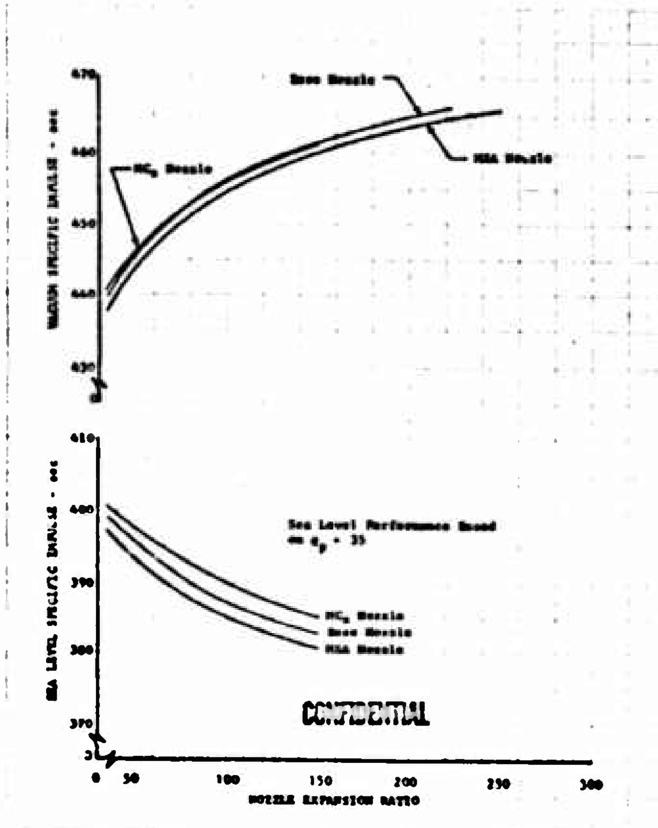


Figure 250. Sea Level and Vacuum Specific Impulse, 250K Common Module, Mixture Ratio of 6.0 DF 59947

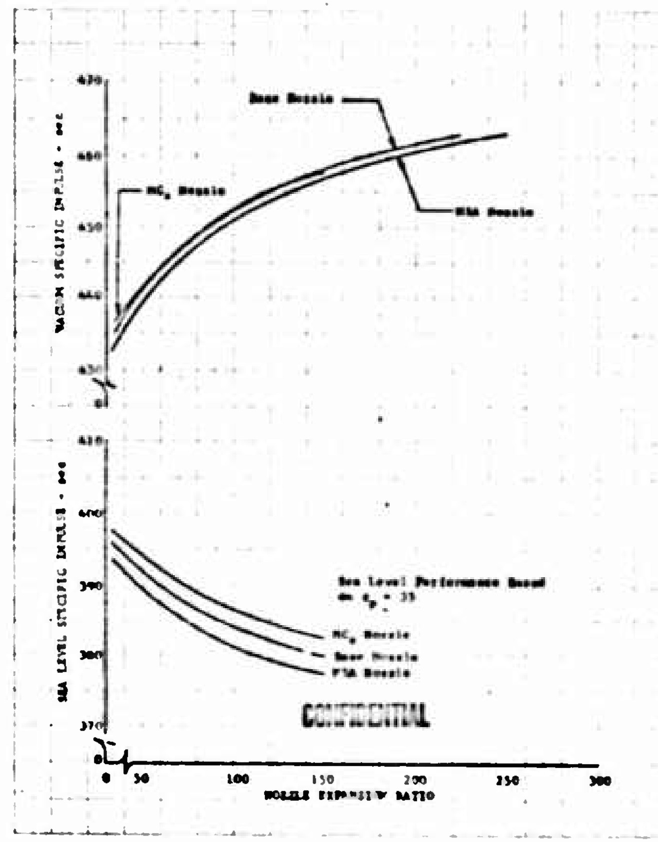


Figure 251. Sea Level and Vacuum Specific Impulse, 250K Common Module, Mixture Ratio of 6.5 DF 59948

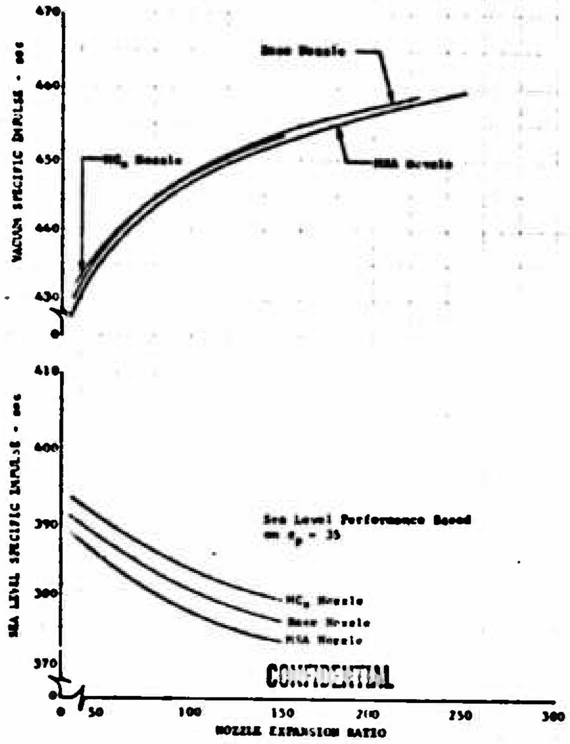


Figure 252. Sea Level and Vacuum Specific Impulse, 250K Common Module, Mixture Ratio of 7.0 DF 59949

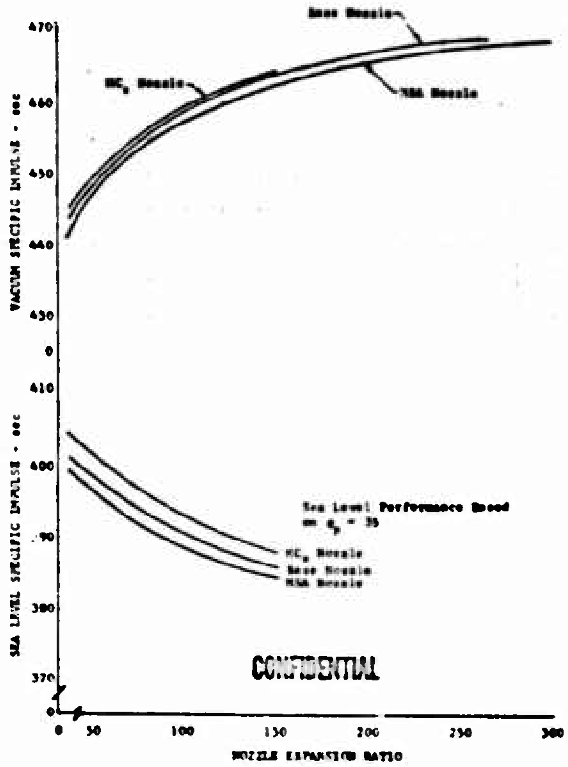


Figure 253. Sea Level and Vacuum Specific Impulse, 350K Common Module, Mixture Ratio of 5.0 DF 59950

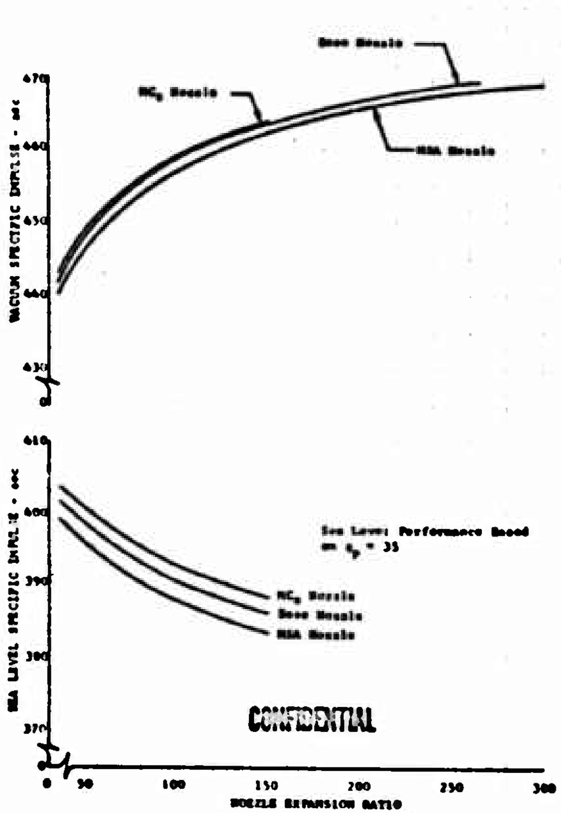


Figure 254. Sea Level and Vacuum Specific Impulse, 350K Common Module, Mixture Ratio of 5.5 DF 59951

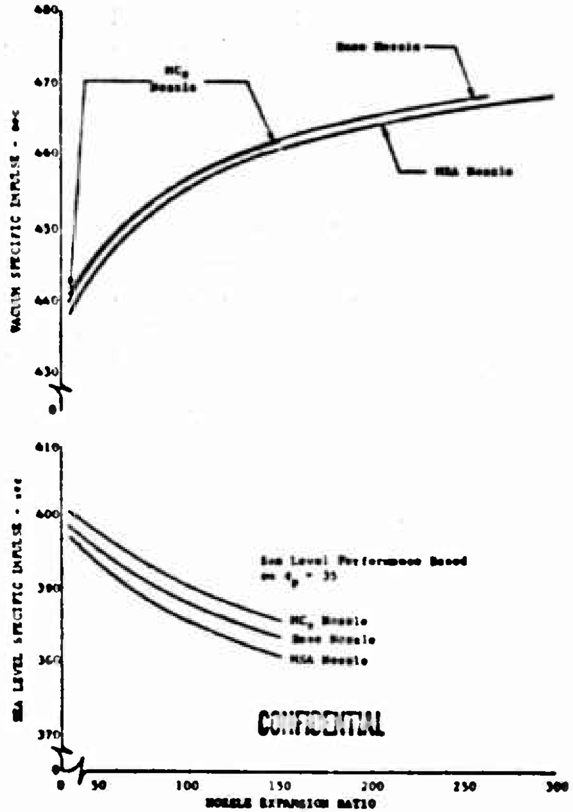


Figure 255. Sea Level and Vacuum Specific Impulse, 350K Common Module, Mixture Ratio of 6.0 DF 59952

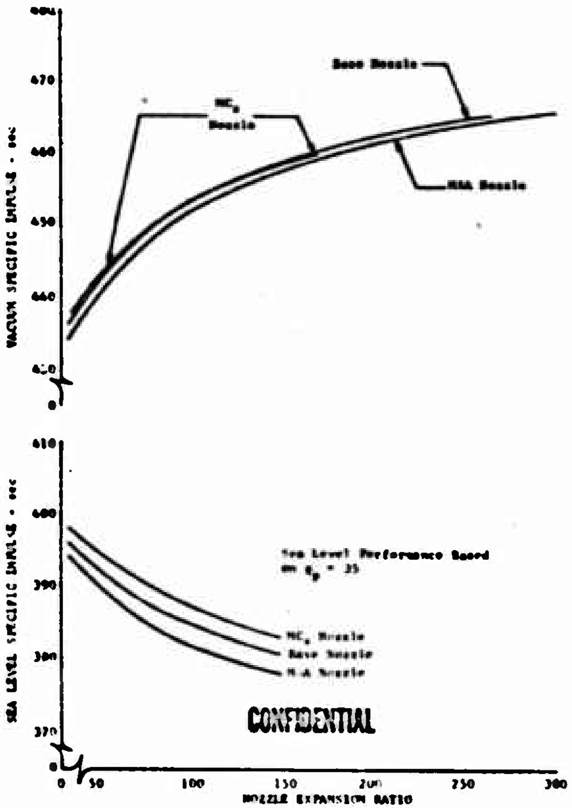


Figure 256. Sea Level and Vacuum Specific Impulse, 350K Common Module, Mixture Ratio of 6.5 DF 59953

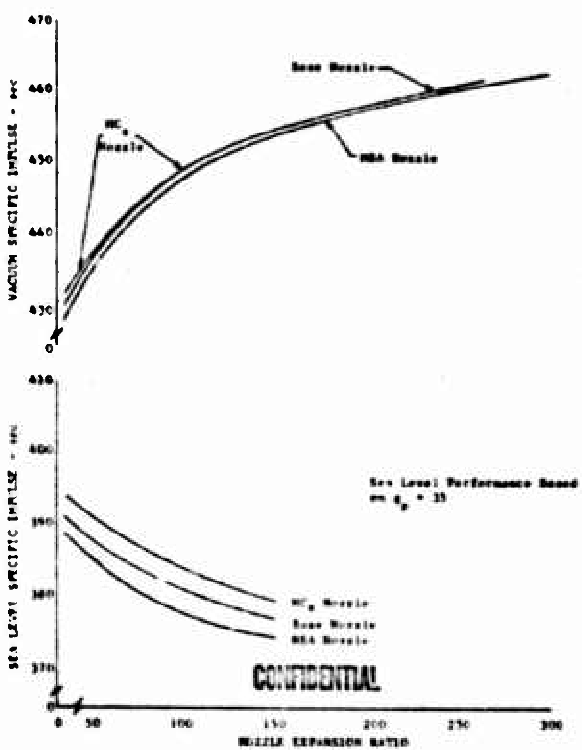


Figure 257. Sea Level and Vacuum Specific Impulse, 350K Common Module, Mixture Ratio of 7.0 DF 59954

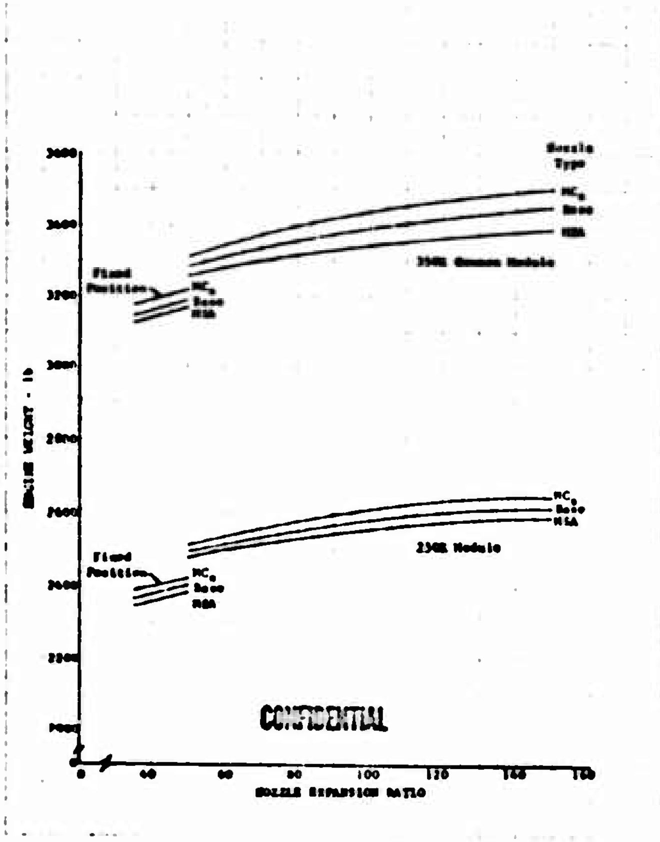


Figure 258. Engine Weight for Common Modules (Nozzles Retract from $\epsilon = 35$) DF 59959

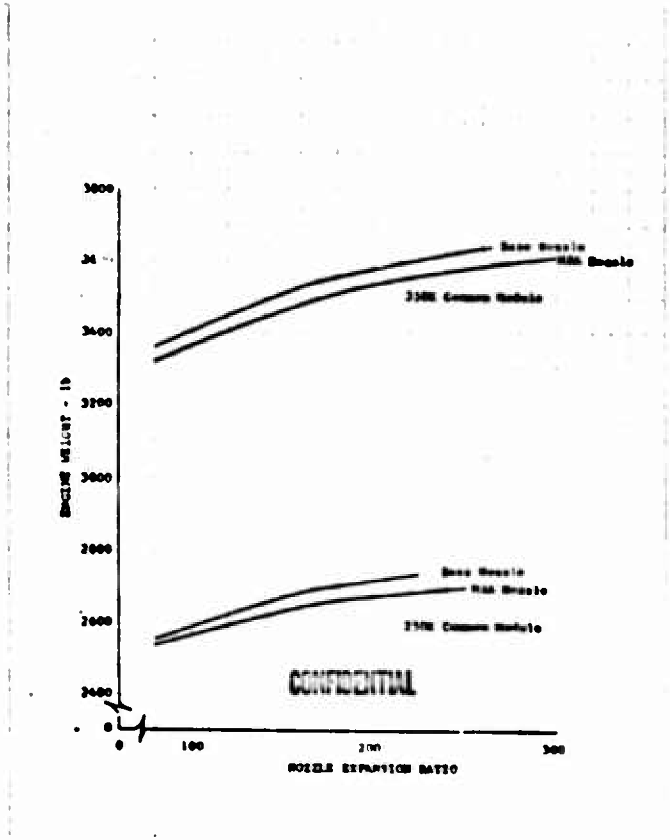


Figure 259. Engine Weight for Common Modules (Nozzles Retract for Minimum Stowed Length) DF 59960

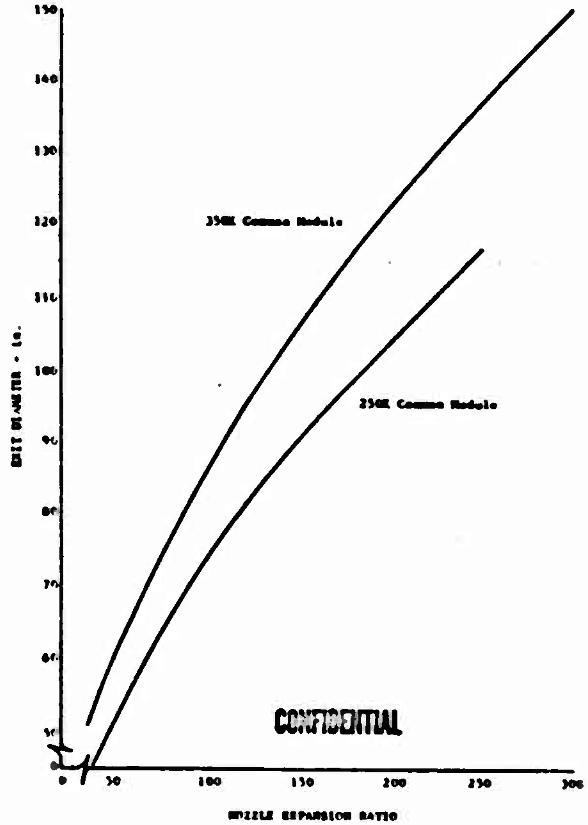


Figure 260. Exit Diameter, Common Module DF 59958

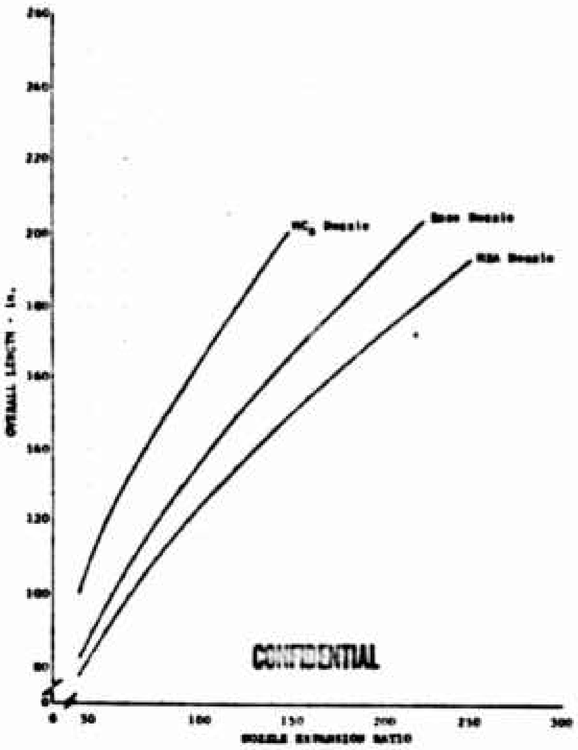


Figure 261. Overall Length of 250K Common Module. DF 59957

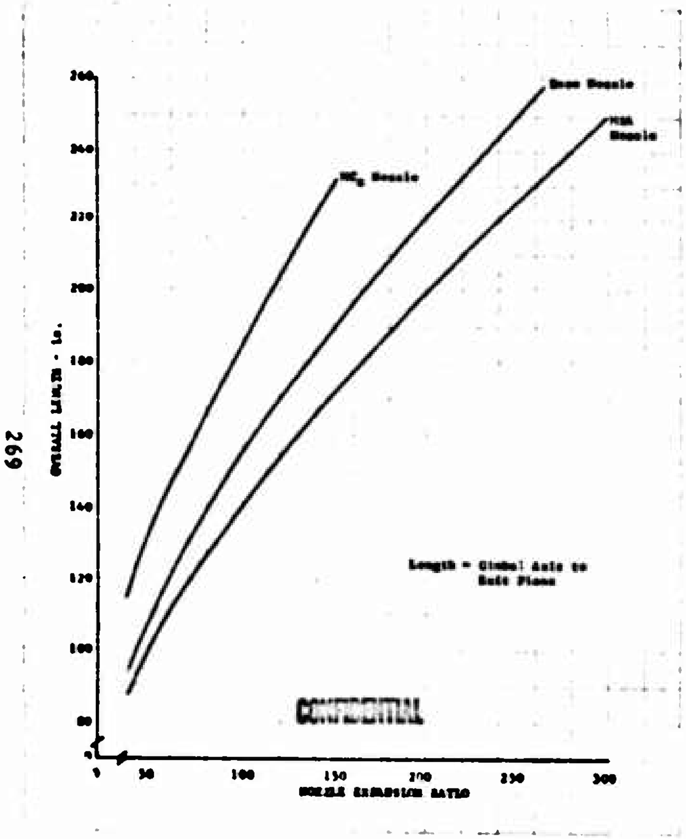


Figure 262. Overall Length of 350K Common Module DF 59963

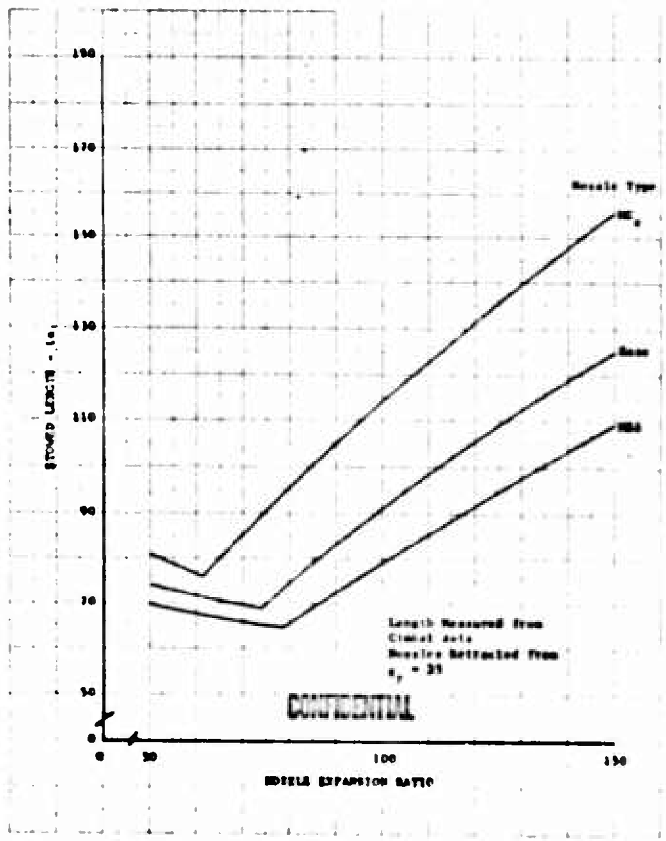


Figure 263. 250K Common Module Stowed Length DF 59955

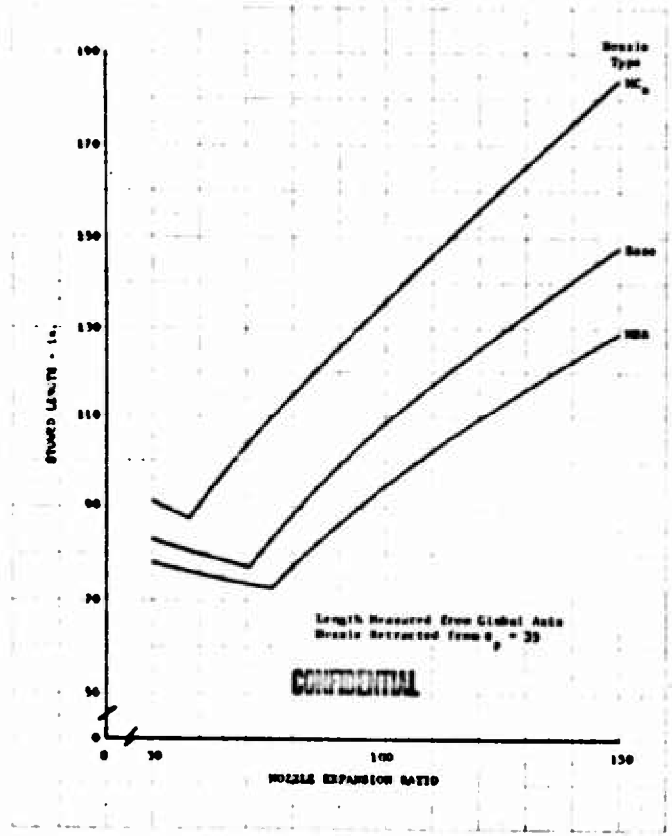


Figure 264. 350K Common Module Stowed Length DF 59961

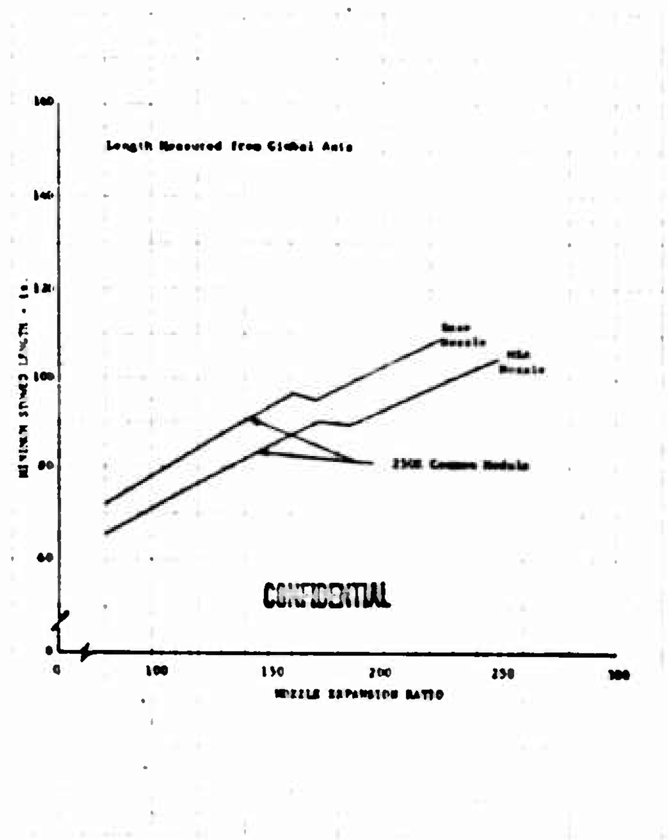


Figure 265. 250K Common Module Minimum Stowed Length DF 59956

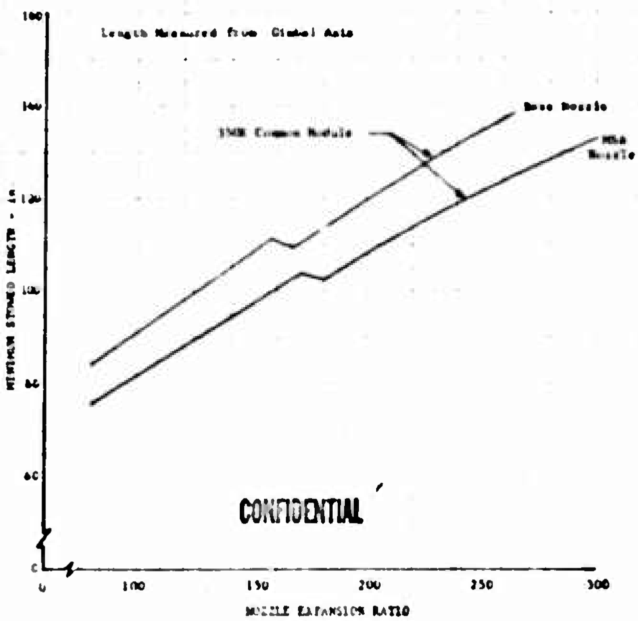


Figure 266. 350K Common Module Minimum Stowed Length DF 59962

APPENDIX IV
APPLICATIONS STUDY GROUND RULES

(U) Specific ground rules applicable to the Applications Study were established by RPL. These ground rules are presented in the following list.

(U) 1. In the 350K Applications Package, an attempt has been made to keep vehicle gross weights similar to the 250K cases, within the limitation of integer numbers of modules.

(U) 2. The dimension given for fairing attachment in Section 3.2.1, Case 3 of the 250K Applications Package should be 60 inches instead of 110 inches.

(U) 3. The Applications Packages were written to allow mixture ratio, O/F, variations during flight to optimize payload.

(U) 4. A vehicle thrust structure spring rate constant of 2×10^6 lb/in. should be used.

(U) 5. The nominal pressures specified at tank exits will remain constant, regardless of vehicle accelerations.

(U) 6. The thrust structure in Case 5 may be extended between the propellant tanks above the mounting plane to allow more compact packaging.

(U) 7. Feed lines between tank exits and engines may be routed above the specified exits if desired.

(U) 8. Heat shielding is required for vehicle protection and may not be necessary for all cases.

(U) 9. The 500°F temperature limit for engine surfaces may be achieved in the most advantageous method, including the localized application of appropriate insulation.

(U) 10. To size the engine for the application study, the upper stage cases will be used to determine the optimum large area ratio nozzle. In this optimization, mixture ratio will also be varied. The optimum large area ratio will be used to size pumps, throat and flow rates at a mixture of 6:1 for 250K of vacuum thrust.

(U) 11. The ratio of maximum side load to axial thrust presented in table 4.2.2 of the applications package will be used to determine maximum deflection for mechanical gimballing to size secondary injection systems. The impulse ratio in table 4.2.2 indicates the influence of TVC on effective thrust, but, it and the ratio of average side load to axial thrust will not be used in this study.

(U) 12. The placement of boost pumps in relation to the engine may be varied depending on the specific application.

(U) 13. The engine in five engine propulsion systems should be placed in a cruciform array.

(U) 14. Application analysis of secondary injection TVC should consider the weight of unexpended injectant, tank weight assignable to the injectant and the weight of secondary injectant system. Expenditure of injectant should be determined by assuming a constant flow rate based on the ratio of average side force to maximum axial thrust. This constant injectant flow rate should start at the beginning of the trajectory and continue for a time sufficient to satisfy the ratio of total side impulse to total axial impulse. An adequate reserve of injectant, consistant with existing practice should be provided. The influence of TVC on effective engine impulse should be neglected.

(U) 15. Heat exchanger weight estimates will be based on maintaining ullage pressures equal to the values specified as nominal tank exit pressures and with collapse factors of 3 for LH_2 pressurant and 2 for LO_2 pressurant.

(U) 16. Feed system weight estimates for the application study will assume pressure drops from tank exits to engine inlets of 3.0 psi for liquid hydrogen and 2.5 psi for liquid oxygen. Prevalves will not be included in these line weights. Symmetrical tank outlet locations will be assumed for Cases 1 and 3.

(U) 17. All engine arrangements in the vehicles of the application study will be symmetric about the pitch axis and symmetric about the yaw axis.

(U) 18. A minimum engine clearance of 2 inches will be used in the application study.

(U) 19. Base heat shield weight estimates for the application study will assume a 1 psi pressure differential (based on vented base data) and a heat shield surface that extends to the skirt.

(U) 20. The maximum dimensions of clustered engine arrangements in the applications study will be less than the diameter of the vehicle skirt.

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13. ABSTRACT

The Applications Study was conducted as a part of the Advanced Development Program for a High Performance Staged Combustion Oxygen/Hydrogen Rocket Engine. The overall objective of the Applications Study was to investigate the application of this engine in six representative advanced rocket stages and to determine the resulting performance. A stage figure of merit (referred to as the Performance Index, W_x) was specified and used for performance evaluation. Performance Index included considerations of engine specific impulse, weight, size, and installation features as these parameters affect vehicle performance. Engine associated weight includes items such as thrust structure, feed lines, thrust vector control, failure detection equipment, and propellant tank pressurization. The analysis was conducted at 250K and 350K vacuum thrust levels in each of six vehicle applications.

Based upon the results of the Applications Study, the lightweight two-position bell nozzle was selected as the basic engine nozzle configuration. Engine nozzle expansion ratio varied considerably in each of the six vehicle cases. Lower stages tended to optimize with lower expansion ratio nozzles and upper stages with higher expansion ratios. A common engine module size for both the 250K and 350K thrust levels was determined by collectively considering the requirements for all six vehicles. Final Performance Index values were determined using this common engine module and exhaust nozzles individually selected (expansion ratio, contour, fixed or two-position) to provide best vehicle performance. The final summed performance of the six vehicle cases (250K engine) was 98.2% of the performance that could be realized for cases where the engine module was individually sized in each vehicle case.

A complete engine description including parametric data and operating parameters is included in the report.

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14. KEY WORDS	LINK A ROLE	LINK A WT	LINK B ROLE	LINK B WT	LINK C ROLE	LINK C WT
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Propulsion Performance						
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